MASSACHUSETTS INSTITUTE OF TECHNOLOGY

APOLLO

GUIDANCE, NAVIGATION AND CONTROL

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R-567

GUIDANCE SYSTEM OPERATIONS PLAN FOR MANNED LM EARTH ORBITAL AND LUNAR MISSIONS USING PROGRAM LUMINARY 1E

SECTION 5 GUIDANCE EQUATIONS (Rev. 11)

DECEMBER 1971

MIT CHARLES STARK DRAPER LABORATORY
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R-567

GUIDANCE SYSTEM OPERATIONS PLAN FOR MANNED LM EARTH ORBITAL AND LUNAR MISSIONS USING PROGRAM LUMINARY 1E

SECTION 5 GUIDANCE EQUATIONS

Signatures appearing on this page designate approval of this document by NASA/MSC.

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FOREWORD

SECTION 5 REVISION 11

The Guidance System Operations Plan (GSOP) for Program LUMINARY is published in six sections as separate volumes.

1. Prelaunch
2. Data Links
3. Digital Autopilot
4. Operational Modes
5. Guidance Equations
7. Erasable Memory Programs

With this issue, Section 5 is revised from the previous issue of LUMINARY GSOP (Revision 10, June 1971) in order to reflect the NASA-MSC-approved changes listed on the revision index cover sheet dated December 1971. The front matter has been re-arranged such that the revision index cover sheets now follow the Foreword and are, in turn, followed by the Table of Contents.

Although the GSOP specifies an earth-orbital capability and this capability has been provided—verification testing shall not be accomplished for earth-orbit rendezvous.

This volume is published as a control document governing the basic design for the guidance and navigation computations of LUMINARY 1E. Revisions constituting changes to the LUMINARY Program require NASA approval.

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REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 0)

PCR (PCN*)	TITLE
7	IMU Alignment Program Change
8	Mod. to ΔV Monitor
15	S-Band Antenna
30	DPS Trimming Time
37	Lambert Fix
39	Z Axis Track
*80.2	State Vector Synchronization

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 1) (Sheet 1 of 3)

PCR (PCN*)	TITLE
13	Mode II Rendezvous Radar Designate
70	Changes to P12, P70, P71
88.3	Erasable Addition
95.2	Sign Change to Noun 72
97.2	RR Monitor Routine Change
105	Deletion of LM/CSM Separation Monitor
106	Deletion of Direct Transfer Ascent Targeting
118	Redefinition of R12 Proceed & Inhibit
121.2	New Engine Fail Routine
122.2	Terminate Ullage at Fixed Time after Engine on Signal
*124.2	Attitude Maneuver During RR Search Routine
133	LM Abort Insertion Targets
134.2	Pulse Torquing to Achieve IMU Realignment
137	R 12 Flag Change
138	Deletion of Predicted Launch Time Program
*144.2	Reduction of Ullage Duration of Delta V Threshold
*146.2	Emergency Termination of Integration Function
164.3	Mode II Attitude Error Display
*171.2	Targeting Interface with P40/41
*173.3	RMS Position and Velocity Error Display
*181.2	Reduction of Sun Occulation Cone
*191.2	Correct GSOP Reference to Range Low Scale Discrete

PCR (PCN*)	TITLE
210	Increase DPS Throttle Recovery Limit
214	DSKY Light Utilization For LR
216	Initiation of LR Antenna Position Change
229	Addition of R77 to Luminary
233	Change to CDH Time Test
244	Delay use of LR Data
246	Implementation of One-Phase Descent Guidance Logic
248	LR Reasonability Test
252	Update GSOP Section 5
253	Landing Radar Read Initiation
400	Provide RR Downlink Data on Lunar Surface in P22
*409.2	Transfer of RR From Auto Track to LGC Mode
*410.2	Prevent Problems Arising from RR Control Mode Changes
*415.2	Prevent Display Conflicts When RR Goes Out of Auto Mode
*431.2	R24/ R61 Interface
*432.2	GSOP Section 4 R21 Repositioning Check
437	MIDTOAVE For P47, P12, P63
439.2	Downgrade the Authority of the Preferred Attitude Flag
*457.2	Correction to R22
468.2	Change R32 into Program P76
470	Addition of P68 Program
471	Revision of RR Remode Release Angles
472	Simplification of P71
*476	FINDCDUW - Gimbal Drive
478	Ascent Guidance Equation Compensation
489	Bypass R-54 and Noun 93 During Initial Alignment
*497	Do Not Delay P63 Throttle-Up Time
*499	R10 Computation Frequency
*507.2	Termination of Integration

PCR (PCN*)	TITLE
527	Mark Verb for R59
537	Surface Navigation Flag Check in P20 and P22
541	Decrease Frequency of Marks in P22
549	RR Shaft/Trunnion Bias Mod. by Crew
*561 †	Reset RENDWFLG in P12
*564	Luminary GSOP Section 5 Update
*565	GSOP Section 5 Control Data Reference Update
568	Delete Use of R29 During P63
*573.3	Definition of A_X and A_Y in Section 5 of GSOP
587.2	APS and DPS TAIL OFF Constants
*604	Provide Maximum Display for Pericenter and Apocenter in P30, P31, R30
608	Simplification of Preferred Orientation Selection (P57)
609	Preferred IMU Orientation When Thrust is Along Local Vertical

† Additional material added in Revision 7.

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 2)

PCR (PCN*)	TITLE
99.2	DSKY Display of RR Position in Mode II
254.2	Modification of CDH Time Computation Logic
258	Redefinition of Vertical Rise Velocity Cutoff
259	Omit Zone 1 from Descent Logic
*564	Luminary GSOP Section 5 Update
576.2	Removal of Backward Updating Constraint on State Vector
609	Preferred IMU Orientation When Thrust is Along Local Vertical
612.2	Raise Thresholds for Delta V Monitor
613	Automatic 4-jet Translation Capability in P12, P70, P71
630	Update of Ascent Guidance Engine Parameters
631	Reduce LGC Executive and Central Processor Load
632	Allow Astronaut to Continue Landing Display, When Radar Does Not Achieve Position #2
*634	Correct Design Flaw in R61, R65 for High LOS Rates
*635	FINDCDUW Gain Change for CSM-Docked Burns
636	Nov. 68 GSOP Section 5 Corrections in Landing Program
639	Altitude Reasonability Test Parameters in Erasable Memory
*640	Remove the Instabilities and Excessive Overshoots from the RR Designate Routine R21
652	Correction to Section 5 APS Minimum Impulse Burn Parameters
*750	Luminary GSOP Section 5 Update (Rev. 2)

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 3)

PCR (PCN*)	TITLE
*758 †	Allow 15 Seconds for an Integration Time Step in R-41
761.1	R-2 Lunar Potential Model

†Additional material added in Revision 7.

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 4)(Sheet 1 of 2)

PCR (PCN*)	TITLE
254.2	Modification of CDH Time Computation Logic
260	Preferred Orientation During LM Aborts
268.2 †	Reduction of P34/35 Run Time
270	Placement of Desired Insertion Radial Velocity Component Into Erasable for P70/P71/P12
*622 †	Correction to R-31 When LM is on the Lunar Surface
636	November, 1968 GSOP Section 5 Corrections in Landing Program
642 †	Provide "Wings Level", Heads Up, Fine Z Axis Tracking
646	Give Astronaut the Option to Confirm Mainlobe Lock-on after R-21 Acquisition
647 †	Replace Lambert with "A" Steer in P40, P41 and P42
654 †	Lessen Delays in R-31
*659.2 †	Suppression of X-Modulo-ing by Kepler
696	V06N22 Display in P57
697	Limitation of LM Abort Orbit Insertion to $1/2^{\circ}$ Plane Change
698 †	Add LM Position Determination Capability to P57
699	Pad Load AOT Back Detent AZ and EL Angles
702	Add COAS Calibration Option to R52
707.2 †	Change from 1968/1969 Ephemeris Data to 1969/1970 Ephemeris Data
708	Provide Continuously Variable Abort Orbit Insertion Targeting
709	Improve TGO Prediction for Short Burns in the BURNTIME Routine

†Additional material added in Revision 7.

Section 5, Revision 4
(Sheet 2 of 2)

PCR (PCN*)	TITLE
716	Ascent Powered Flight RCS Control
719	Speed Up P21
720	Abort Coasting Integration When in Infinite Acceleration Overflow Loop
721	Time-Theta and Time-Radius Alarm Abort
722	Improve Performance of RR Designate Procedure on Lunar Surface
732 †	Permit the Crew to Modify W-Matrix Bias Error in V67 Routine
736	Add Source Code to Noun 49 in P20/P22
738	H, V, γ Display with P21
*744	Change ϵ to 1.5 Seconds in R24
754	Provide IMU Orientation Selection Option Code in P57
*755	Change IMU Gimbal Angles in Gravity Vector Determination

†Additional material added in Revision 7.

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GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 5)(Sheet 1 of 2)

PCR (PCN*)	TITLE
279	Variable Insertion Computation with Capability to Abort at any Time
670	Simplification of Landing Programs
688	Guidance Frame Erection Check
695 †	Provide option for CSI Program to compute T(APOAPSIS)
700A	Improve the Rate of Descent Mode (P66) Performance
723	Two-Segment LR Altitude and Velocity Weighting Functions
*731	Modify the Lunar Landing Guidance Equations to Compensate for Computation, Throttle, FINDCDUW, and Attitude Control Lags
737	Permit ATT HOLD Mode in P63, 64, 65
751	Make 1406 Alarm Non-Abortive
756	Guidance Frame Erection Trajectory Shaping Factors
762	Delete V68
*765 †	New Propulsion System Constants
772.2 †	Libration Vector at Landing Time
773.2	Fix Constants for Planetary Inertial Orientation Subroutine
775	Modify R12 to Permit LGC Compensation for Doppler in LR Range Reading
776.2	Improved R2 Model Timing
780 †	Provide Pure RR Range, Range Rate, and Time Tag during P20, P22 and P25.

† Additional material added in a later revision.

PCR (PCN*)	TITLE
801.2†	Make BAILOUT Alarms Start with 3XXXX and POODOO Alarms Start with 2XXXX.
817	Eliminate Undesirable LR Position Alarms from R12
818†	Permit Rejection of Individual Measurement Incorporations in P20
820	Eliminate Lighting of ALT Light when Low Scale Discrete is Absent
823†	Delete P31 (Lambert Aim Point Guidance Program)
*830	Supplementary ASTEER Modifications
*831.2	Lambert Overflow Protection
832.2†	Remove Restriction of Running R05 Only in POO
834	Descent Guidance Corrections
839†	R12 and LR Re-position Routine Improvements
840†	Reduce Attitude Oscillations in P64 and P65
844†	Deletion of P38/P78 and P39/P79
845†	Do Not Turn On R29 During P70/P71
847	Eliminate Possible Lock-out of Pitch-over from P12, P70, P71

†Additional material added in Revision 7.

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY

Section No. 5 Title: Guidance Equations (Revision 6)

PCR (PCN*)	TITLE
848	Prevent RR ECDUs from Stealing LGC Memory Cycles
854 †	Provide a Flexible Method for Crew to Modify RLS
855	Begin Reading LR Velocity as soon as Velocity Data Good Appears

†Additional material added in Revision 7.

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY 1C

Section No. 5 Title: Guidance Equations (Revision 7)

PCR (PCN*)	TITLE
285	Remove check of Auto Throttle Discrete
846, REV 1	More Accurate Delta T Tail-Off for P70
863.2	Make P76 Set NODO Flag
882	Replace VHORIZ with Something Better
893	Abort Targeting Flagbit
895	LR Reposition by V59E in P63
936.2	Initialize V90 time to TIG
943	Velocity Reasonability Test
968	LPD Bias Correction
971	Change Fixed Memory Constant (APS) Delta-T TAIL-OFF
972	Display Polarity of Sighting Angle Difference in R54

Date: April 1970

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GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program LUMINARY 1C (LM131 Rev 1)

Section No. 5 Title: Guidance Equations (Revision 8)

Revision 8 is published as change pages to Section 5 LUMINARY GSOP. Substitution of these pages for those in Revision 7 makes Section 5 the control document for guidance equations for the re-release of Program LUMINARY 1C (LM131 Rev 1). The following NASA/MSC approved changes are included in Revision 8:

PCR (PCN*)	TITLE
942	LR Update Cut off
988	Auto P66
1013	Multiple Servicers Avoidance in P66
1033*	Section 5, LUMINARY 1C GSOP Changes

Date: December 1970

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program Luminary 1D

Section No. 5 Title: Guidance Equations (Revision 9)(Sheet 1 of 2)

Revision 9 incorporates the following NASA/MSC approved changes and becomes the control document for LUMINARY 1D (Rev. 178).

PCR (PCN*)	Title
287	Removal of 526 Alarm in P22 and P20
296	Set "G" Vector Parallel to Landing Site Radius Vector
298	Decrease Time to Call Alarm Code 523
310	Time to Call 511 Alarms
322	Change Fixed Constant "Throtlag" from 0.2 to 0.08
821.2	Move AZO to Fixed Memory
892	Delete R29
896	LR Velocity Read Centered at PIPTIME
897	Delete PCR 775
898	LR Velocity Read
979	Delete 521 Alarm
982	Extend Capability of Lunar Surface Star Acquisition Routine R59
986.2	Update Fixed Constants for 1970 – 1971 Ephemeris Year
1010*	Sect. 5 Rev. 9 GSOP Changes
1021†	Fixed Memory Landing Radar Transformation Matrices
1022	Landing Radar Position Alarms
1025	Remove Gravity Computation After Landing Radar Altitude Update
1027	A priori Terrain Models
1028	Two-segment Altitude Weighting Functions for Landing Maneuver

†Additional material added in Revision 10

(PCN*)	Title
1035*†	V68 and P66 Terminate the Terrain Model
1037*†	P66 Corrections
1038	Keep 526 Alarm in P20 (PCR 287)
1039*	TERRAIN Model Improvement (PCR 1027)
1052*	P66 IMU/c.g. Offset Compensation
1056 Rev. 1	Improvements for Impulse and Ullage Logic
1058	New Landing Analog Displays (R10)
1069.2*	Delete Rendezvous Test for Earth Orbit

†Additional material added in Revision 10.

Date: June 1971

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program Luminary 1E

Section No. 5 Title: Guidance Equations (Revision 10)

Revision 10 incorporates the following NASA/MSC approved changes and becomes the control document for LUMINARY 1E (Rev. 210)

PCR (PCN*)	Title
319	A Priori Terrain
324	PGNCS/AGS RR Data Transfer
338	Change LPD Scaling to 1.0° In All Directions
341	Landing Radar Reasonability Test
348	New Target - ΔV Program for LGC
1044	Re-design of R53-R57
1066	Display N81 On All Passes In P34, P35, P74, P75.
1082.2	Update Fixed Constants for 1971-1972 Ephemeris Year
1107	Back up of Abort Bits Channel 30 Bits 1 and 4 for On (0)
1109	Back up of Off (1) Failures of Auto Throttle Channel 30 Bit 5
1110	Back up for Failures of the Display Inertial Data Bit Channel 30 Bit 6
1126*	Set NOTHROTL for DPS Impulse Burns
1134	Revision of PCR 1111: Back-up of Guid Select and Mode Control Switches
1141*	Initialize FLRCS in P12
1150*	Section 5 Revision 10 GSOP Changes

Date: December 1971

REVISION INDEX COVER SHEET

GUIDANCE SYSTEM OPERATIONS PLAN

GSOP No. R-567 Title: For Manned LM Earth Orbital and Lunar Missions Using Program Luminary 1E

Section No. 5 Title: Guidance Equations (Revision 11)

Revision 11 incorporates the following NASA/MSC approved changes. These changes are indicated by a series of bars in the margin.

PCR (PCN*)	Title	Pages Affected
334R1	Change DSKY DESCENT/ASCENT NOUNS	5.3-89, 118, 131, 143, 151
1137*	Correct constant for ascent guidance.	5.3-136

Also included are several document-improvement changes indicated by a series of dots in the margin as authorized by PCN 1181. The signatures on page iii indicate NASA approval of these changes.

Revision 11 becomes the control document for Program Luminary 1E (Rev. 210).

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TABLE OF CONTENTS

SECTION 5

GUIDANCE EQUATIONS

5.1 INTRODUCTION

5.1.1 GENERAL COMMENTS

The purpose of this section is to present the Guidance and Navigation Computation Routines associated with the LM Apollo Lunar Landing Mission. These Routines are utilized by the Programs outlined in Section 4 where astronaut and other subsystem interface and operational requirements are described. The guidance and navigation equations and procedures presented in Section 5 are primarily concerned with mission type programs representing the current LM PGNCS Computer (LGC) capability. A restricted number of LGC service type program operations which involve computation requirements are also included.

The LM PGNCS Computer (LGC) guidance and navigation equations for the lunar landing mission are presented in the following five catagories:

Section 5.2	Coasting Flight Navigation Routines
Section 5.3	Powered Flight Navigation and Guidance Routines
Section 5.4	Targeting Routines
Section 5.5	Basic Subroutines
Section 5.6	General Service Routines

Guidance equation parameters required for program initialization and operation are listed in Section 5.7. These selected parameters are stored in the LGC erasable memory. General constants used in the equations of this volume are presented in Section 5.8.

A cross-reference between the LGC programs and routines of Section 4 that are described in Section 5 is listed in Section 5.1.2. In the Section 5 table of contents and text, missing section numbers correspond to LUMINARY programs that have been deleted from the previous Section 5 GSOPs by MSC direction resulting from the LGC Fixed Memory Storage Review Meeting of August 28, 1967 and subsequent PCN's and PCR's.

5.1.2 Sections 4 and 5 Cross Reference

PROGRAM NUMBER	SECTION 4 TITLE	PRINCIPAL SECTION 5 SUBSECTION NO.	PAGE
P-00	LGC Idling Program	5.6.11	5.6-72
P-12	Powered Ascent Program	5.3.5	5.3-125
P-20	Rendezvous Navigation Program	5.2.4	5.2-35
P-21	Ground Track Determination Program	5.6.5	5.6-59
P-22	Lunar Surface Navigation Program	5.2.5	5.2-81
P-25	Preferred Tracking Attitude Program	5.2.4.4	5.2-77
P-30	External Delta V Program	5.3.3.3.1	5.3-19
P-32	Co-Elliptic Sequence Initiation (CSI) Program	5.4.2.2	5.4-6
P-33	Constant Delta Altitude (CDH) Program	5.4.2.3	5.4-13
P-34	Transfer Phase Initiation (TPI) Program	5.4.2.4	5.4-22
P-35	Transfer Phase Midcourse (TPM) Program	5.4.2.5	5.4-28

PROGRAM NUMBER	SECTION 4 TITLE	PRINCIPAL SECTION 5 SUBSECTION NO.	PAGE
P-40	DPS Program	5.3.3	5.3-8
P-41	RCS Program	5.3.3	5.3-8
P-42	APS Program	5.3.3	5.3-8
P-47	Thrust Monitor Program	5.3.6	5.3-155
P-51	IMU Orientation Determination	5.6.2.1.1	5.6-2
P-52	IMU Realign Program	5.6.2.1.2	5.6-3
P-57	Lunar Surface Align Program	5.6.2.2	5.6-10
P-63	Braking Phase Program	5.3.4.6	5.3-83
P-64	Approach Phase Program		
P-66	Landing Phase Program		
P-68	Landing Confirmation Program	5.3.4.9	5.3-121
P-70	DPS Abort Program	5.4.3.1	5.4-32
P-71	APS Abort Program		
P-72	CSM Co-elliptic Sequence Initiation (CSI) Targeting Program	5.4.2.2	5.4-6
P-73	CSM Constant Delta Altitude (CDH) Targeting Program	5.4.2.3	5.4-13
P-74	CSM Transfer Phase Initiation (TPI) Targeting Program	5.4.2.4	5.4-22
P-75	CSM Transfer Phase Mid-course (TPM) Targeting Program	5.4.2.5	5.4-28
P-76	Target Delta V Program	5.6.16	5.6-88
P-77	Impulsive Delta V Program	5.6.16	5.6-88

ROUTINES	SECTION 4 TITLE	PRINCIPAL SECTION 5 SUBSECTION NO.	PAGE
R-04	RR/LR Self Test Routine	5.6.14	5.6-81
R-05	S-Band Antenna Routine	5.6.6	5.6-60
R-10	Landing Analog Displays Routine	5.3.4.8	5.3-117
R-12	Descent State Vector Update Routine	5.3.4.3	5.3-63
R-13	Landing Auto Modes Monitor Routine	5.3.4.6	5.3-83
R-21	RR Designate Routine	5.2.4.1.1	5.2-39
R-22	RR Data Read Routine	5.2.4.2.1	5.2-53
R-23	RR Manual Acquisition Routine	5.2.4.1.2	5.2-47
R-24	RR Search Routine	5.2.4.1.3	5.2-48
R-25	RR Monitor Routine	5.2.4.3	5.2-75
R-26	Lunar Surface RR Pre—designate Routine	5.2.5.3	5.2-86
R-30	Orbital Parameters Display Routine	5.6.17	5.6-90
R-31	Rendezvous Parameter Display Routine	5.6.7.1	5.6-64
R-36	Rendezvous Out-of-Plane Display Routine	5.6.7.3	5.6-65
R-40	DPS/APS Thrust Fail Routine	5.3.3.6	5.3-44
R-41	State Vector Integration (MID TO AVE) Routine	5.3.8	5.3-174
R-47	AGS Initialization Routine	5.6.8	5.6-68
R-50	Coarse Align Routine	5.6.2.1.2	5.6-3
R-51	Inflight Fine Align	5.6.2.1.2	5.6-3
R-52	Automatic Optics (LM) Positioning Routine	5.6.2.1.2	5.6-3

ROUTINES	SECTION 4 TITLE	PRINCIPAL SECTION 5 SUBSECTION NO.	PAGE
R-53	AOT Mark Routine	5.6.2.1 5.6.3.1.1	5.6-2 5.6-28
R-54	Sighting Data Display Routine	5.6.2.1.1	5.6-2
R-55	Gyro Torquing Routine	5.6.2.1	5.6-2
R-59	Lunar Surface Sighting Mark Routine	5.6.2.2 5.6.3.1.2	5.6-10 5.6-34
R-61	Preferred Tracking Attitude Routine	5.2.4.4	5.2-77
R-63	Rendezvous Final Attitude Routine	5.6.7.2	5.6-65
R-65	Fine Preferred Tracking Attitude Routine	5.2.4.4	5.2-77
R-77	LR Spurious Test Routine	5.6.20	5.6-100

5.1.3 GENERAL PROGRAM UTILIZATION

The following outline is a brief summary of the major LGC programs that would be used in the principal phases of a nominal lunar landing mission. This outline reflects the LGC capability for nominal and abort cases of such a mission.

I Lunar Orbit Phase Prior to Powered Descent Initiation

A) Nominal

Voice Link LGC Initialization

P-27 LGC Update Program (LGC Initialization if required followed by lunar landing targeting and timing parameters)

R-47 AGS Initialization Routine

P-47 Thrust Monitor Program

P-76 Target Delta V Program (CSM circular) (Manual Separation)

P-30 External Delta V Program (No PDI 'Abort targets')

B) Aborts to Return to Earth Trajectory (SPS Backup)

P-47 Thrust Monitor Program (Manual Re-docking)

P-27 LGC Update Program (TEI targeting, state vector updating and cislunar MCC targeting as required)

P-30 External Delta V Program (TEI and cislunar MCC for SPS backup with RTCC targeting)

P-40 DPS Program

I Lunar Orbit Phase Prior to LM Powered Descent Initiation (cont)

C) Service Programs for Nominal and Abort Cases

P - 52	IMU Realign Program
R - 05	S-Band Antenna Routine
P - 21	Ground Track Determination Program
R - 30	Orbital Parameters Display Routine
P - 00	LGC Idling Program
R - 03	DAP Data Load Routine
R - 62	Crew-Defined Maneuver Routine

III Powered Landing Maneuver and Post Landing Phase

A) Nominal

P-63 Braking Phase Program

P-64 Approach Phase Program

P-66 Landing Phase Program

P-68 Landing Confirmation Program

P-57 Lunar Surface Align Program

R-47 AGS Initialization Routine

B) Abort to Orbit

1. Aborts During Powered Landing Maneuver

P-70 DPS Abort Program

P-71 APS Abort Program

2. Aborts from the Lunar Surface (Anytime Launch Case)

P-27 LGC Update Program (CSM State vector update)

P-57 Lunar Surface Align Program (Fast Alignment Mode)

P-12 Powered Ascent Program

C) No-PDI Abort to Rendezvous

P-52 IMU Realign Program

P-30 External Delta V Program

P-40, P-41, P-42 DPS, RCS, or APS Program

IV Lunar Pre-Launch Phase (Final 3 CSM Orbits before LM Launch)

A) Nominal

P - 27	LGC Update Program (state vector updates and launch time)
(or)	
P - 22	Lunar Surface Navigation Program
R - 47	AGS Initialization Routine

B) Service Programs

P - 57	Lunar Surface Align Program
R - 05	S-Band Antenna Routine
P - 00	LGC Idling Program

V LM Powered Ascent Phase

A) Nominal

P - 12 Powered Ascent Program

B) Aborts

None

VI Rendezvous Phase

A) Nominal

P - 20 Rendezvous Navigation Program

P - 34 Transfer Phase Initiation (TPI) Program

P - 35 **Transfer Phase Midcourse (TPM) Program**

P - 41 } RCS or APS Programs
P - 42 }

R - 47 AGS Initialization Routine

P - 47 Thrust Monitor Program (Manual Terminal Rendezvous Maneuvers)

VI Rendezvous Phase (cont)

B) Aborts to Rendezvous

1. LM Active (RR Failure)

P - 27 LGC Update Program (state vector updates and rendezvous targeting)

P - 25 Preferred Tracking Attitude Program (CSM navigation)

P - 30 External Delta V Program (CSM or RTCC targeted)

P - 41
P - 42 } RCS or APS Programs

P - 47 Thrust Monitor Program (Manual Terminal Rendezvous Maneuvers)

2. CSM Active Retrieval

P - 20 Rendezvous Navigation Program (LGC navigation)

P - 76 Target Delta V Program

P - 72 CSM Co-elliptic Sequence Initiation (CSI) Targeting Program

P - 73 CSM Constant Delta Altitude (CDH) Targeting Program

P - 74 CSM Transfer Phase Initiation (TPI) Targeting Program

P - 75 CSM Transfer Phase Midcourse (TPM) Targeting Program

P - 25 Preferred Tracking Attitude Program (CMC navigation)

VI Rendezvous Phase (cont)

C) Service Programs for Nominal and Abort Cases

P - 52	IMU Realign Program
R - 05	S-Band Antenna Routine
R - 30	Orbital Parameters Display Routine
R - 31	Rendezvous Parameter Display Routine
R - 36	Rendezvous Out-of-Plane Display Routine
R - 63	Rendezvous Final Attitude Routine
P - 21	Ground Track Determination Program
P - 00	LGC Idling Program

5.1.4 COORDINATE SYSTEMS

There are six major coordinate systems used in the navigation and guidance programs. These six coordinate systems are defined individually in the following descriptions, and referenced to control specifications of Section 5.8.2 where applicable. Any other coordinate system used in any particular LGC program is defined in the individual section describing that program.

5.1.4.1 Basic Reference Coordinate System

The Basic Reference Coordinate System is an orthogonal inertial coordinate system whose origin is located at either the moon or the earth center of mass. The orientation of this coordinate system is defined by the line of intersection of the mean earth equatorial plane and the mean orbit of the earth (the ecliptic) at the beginning of the Besselian year which starts Jan. 1.2516251, 1972 E.T. The X-axis ($\underline{u}_{XI}$) is along this intersection with the positive sense in the direction of the ascending node of the ecliptic on the equator (the equinox), the Z-axis ($\underline{u}_{ZI}$) is along the mean earth north pole, and the Y-axis ($\underline{u}_{YI}$) completes the right-handed triad. In the lunar landing mission this coordinate system is located at the moon center of mass. During earth-orbital missions in which the LM is active near the earth, the Basic Reference Coordinate System will be earth-centered. All navigation stars and lunar-solar ephemerides are referenced to this coordinate system. All vehicle state vectors are referenced to this system during coasting or free fall phases of the mission.

The Basic Reference Coordinate System is presented in Ref. 1 of Section 5.8.2 as Standard Coordinate System 4, Geocentric Inertial.

5.1.4.2 IMU Stable Member or Platform Coordinate System

The orthogonal inertial coordinate system defined by the PGNCS inertial measurement unit (IMU) is dependent upon the current IMU alignment. There are many possible alignments during a mission, but the primary IMU alignment orientations described in Section 5.6.3.4 are summarized below and designated by the subscript SM:

1. Preferred Alignment

$$\underline{u}_{XSM} = \underline{u}_{TD}$$

$$\underline{u}_{YSM} = \begin{cases} \text{UNIT} (\underline{u}_{XSM} \times \underline{r}) & \text{If } \underline{r} \text{ not parallel to } \underline{u}_{TD} \\ \text{UNIT} (\underline{u}_{XSM} \times \underline{v}) & \text{If } \underline{r} \text{ parallel to } \underline{u}_{TD} \end{cases}$$

$$\underline{u}_{ZSM} = \underline{u}_{XSM} \times \underline{u}_{YSM} \tag{5.1.1}$$

where:

$\left.\begin{matrix} \underline{u}_{XSM} \\ \underline{u}_{YSM} \\ \underline{u}_{ZSM} \end{matrix}\right\}$ IMU stable member coordinate unit vectors referenced to the Basic Reference Coordinate System

$\underline{u}_{TD}$ = unit vector in desired thrust direction at ignition

$\underline{r}$ = position vector at ignition

$\underline{v}$ = velocity vector at ignition

2. Nominal Alignment (Local Vertical)

$$\underline{u}_{XSM} = \text{UNIT} (\underline{r}) \text{ at } t_{align}$$

$$\underline{u}_{YSM} = \text{UNIT} (\underline{v} \times \underline{r}) \tag{5.1.2}$$

$$\underline{u}_{ZSM} = \underline{u}_{XSM} \times \underline{u}_{YSM}$$

where $\underline{r}$ and $\underline{v}$ represent the vehicle state vector at the alignment time, t_{align}.

3. Lunar Landing Alignment

$$\underline{u}_{XSM} = \text{UNIT}\,(\underline{r}_{LS})\ \text{at}\ t_L$$

$$\underline{u}_{ZSM} = \text{UNIT}\left[(\underline{r}_C \times \underline{v}_C) \times \underline{u}_{XSM}\right] \quad (5.1.3)$$

$$\underline{u}_{YSM} = \underline{u}_{ZSM} \times \underline{u}_{XSM}$$

where $\underline{r}_{LS}$ is the lunar landing site vector at the predicted landing time, t_L, and $\underline{r}_C$ and $\underline{v}_C$ are the CSM position and velocity vectors, as maintained in the LGC.

4. Lunar Launch Alignment

The same as that defined in Eq. (5.1.3) except that $\underline{r}_{LS}$ is the landing or launch site at the predicted launch time t_L.

The origin of the IMU Stable Member Coordinate System is nominally the center of the IMU stable member. In the following programs, however, the origin of the IMU or platform coordinate system is located at the moon center of mass:

Lunar landing programs P-63, P-64, P-66, and P-68
Powered Ascent Program P-12
Abort programs P-70 and P-71

5.1.4.3 Vehicle or Body Coordinate System

The Vehicle or Body Coordinate System is the orthogonal coordinate system used for the LM structural body. The origin of this coordinate system is 200 inches below the LM ascent stage base. The X-axis ($\underline{u}_{XB}$) lies along the longitudinal axis (centerline of the transfer tunnel) of the LM, positive in the nominal DPS-APS thrust direction. The Z-axis ($\underline{u}_{ZB}$) is parallel to the centerline of the exit hatch and directed forward from the design eye. The Y-axis ($\underline{u}_{YB}$) completes the right-handed triad. This coordinate system is defined in Ref. 1 of Section 5.8.2 as Standard Coordinate System 8d, LEM Structural Body Axis.

5.1.4.4 Earth-Fixed Coordinate System

The Earth-Fixed Coordinate System is an orthogonal rotating coordinate system whose origin is at the center of mass of the earth. This coordinate system is shown in Ref. 1 of Sec. 5.8.2 as the Standard Coordinate System 1, Geographic Polar. The Z-axis of this coordinate system is defined to be along the earth's true rotational or polar axis. The X-axis is defined to be along the intersection of the prime (Greenwich) meridian and the equatorial plane of the earth, and the Y-axis is in the equatorial plane and completes the right-handed triad.

5.1.4.5 Moon-Fixed Coordinate System

The Moon-Fixed Coordinate System is an orthogonal rotating coordinate system whose origin is at the center of mass of the moon. This coordinate system is shown in Ref. 1 of Sec. 5.8.2 as the Standard Coordinate System 2, Selenographic Polar. The Z-axis is defined to be along the true polar or rotation axis of the moon, the X-axis is through the mean center of the apparent disk or along the intersection of the meridian of 0° longitude and the equatorial plane of the moon, and the Y-axis is in the equatorial plane and completes the right-handed triad.

5.1.4.6 Navigation Base Coordinate System

The Navigation Base Coordinate System is an orthogonal coordinate system whose origin and axis orientation are defined by three mounting points between the PGNCS navigation base and the LM vehicle structure. These mounting point locations are defined in Section G-G of the GAEC-MIT Interface Control Document LID280-10004 (Ref. 13 of Sec. 5.8.2). The Y_{NB} axis is defined by the centers of the two upper mounting points along line F of Ref. 13, with the positive direction in the same general direction as the LM +Y vehicle axis. The X_{NB} axis is defined by a line through the center of the lower mounting point (point K of Ref. 13) and perpendicular to the Y_{NB} axis. The positive X_{NB} direction is in the same general sense as the LM +X vehicle axis. The $+Z_{NB}$ defined as $\underline{X}_{NB} \times \underline{Y}_{NB}$ to complete the right-handed triad. The Navigation Base Coordinate System is approximately parallel with the LM Vehicle Coordinate System.

5.1.5 GENERAL DEFINITIONS AND CONVENTIONS

In this section the definitions of and the conventions for using a selected number of parameters are given. Although virtually all of the information in this section appears elsewhere in this document, this section provides a summary of facts which are distributed among various other sections.

5.1.5.1 Error Transition Matrix Maintenance

5.1.5.1.1 Definitions

The error transition matrix (W matrix) is defined in Section 5.2.2.4 and is used in processing navigation measurement data. Control of the W matrix is maintained by means of the flag RENDWFLG (see Sections 5.2.4.2.2 and 5.2.5.4). If RENDWFLG is equal to one, then the W matrix is valid for processing rendezvous navigation data; while this flag being equal to zero indicates that the W matrix is invalid.

5.1.5.1.2 W Matrix Control Flag

The W matrix control flag is maintained according to the following rules:

1. RENDWFLG is initially zero.

2. A CSM state vector update from the ground (RTCC) causes the flag to be zeroed.

3. A LM in-flight state vector update from the ground causes the flag to be zeroed. An update of the landing site vector when the LM is on the lunar surface does not cause the flag to be zeroed.

4. There exist special DSKY procedures by which the astronaut can zero the flag (verbs 67 and 93).

5. Deduction of liftoff by the lunar ascent program (P12) (or P70, P71 aborts) causes RENDWFLG to be zeroed.

6. Overflow of the W matrix during extrapolation causes RENDWFLG to be zeroed.

7. Initialization of the W matrix for rendezvous or lunar surface navigation causes RENDWFLG to be set to one.

With regard to the last item 7 above, there exist in erasable memory two sets of initialization parameters for the W matrix: one for rendezvous navigation and one for lunar surface navigation. Each of these sets contains two elements, a position element and a velocity element. In addition, the set for rendezvous contains initialization parameters for shaft and trunnion. At the time each set of navigation data is processed, RENDWFLG is tested. If the flag is found to be zero, then the W matrix is initialized consistent with the appropriate erasable parameters, and the flag is set to one. See sections 5.2.4.2.2 and 5.2.5.4 for more complete details of this initialization procedure.

5.1.5.1.3 W Matrix Extrapolation

Extrapolation of the W matrix is described in Section 5.2.2.4. Required in this extrapolation is the specification of the appropriate vehicle's state vector with which the W matrix is extrapolated. This extrapolation occurs during programs P-00, P-20, and P-22; and at the conclusion of programs P-40, P-41, P-42, and P-47. The conventions under which the extrapolation occurs during each of these programs are as follows:

P-00 : If RENDWFLG is equal to one, the W matrix is extrapolated with the LM state vector if the LM is in flight and with the CSM state vector if the LM is on the lunar surface. The W matrix is not extrapolated if the flag is equal to zero. (See Section 5.6.11).

P-20 : The W matrix is extrapolated with the state vector that is being updated if RENDWFLG is equal to one, and not extrapolated if RENDWFLG is equal to zero. (See Section 5.2.4.2.2.)

P-22 : The W matrix is extrapolated with the CSM state vector if RENDWFLG is equal to 1, and not extrapolated if RENDWFLG is equal to zero. (See Section 5.2.5.4.)

P-40, P-41, P-42, P-47 : The result of the maneuver will be a final state vector at the end-of-maneuver time t_F. The LM state vector that existed before the maneuver program will still exist; and cotemporal with it, there will also be the CSM state vector and the W matrix. The following steps are performed before the program is terminated:

1. If the W matrix control flag is equal to one, the old LM state vector and the W matrix are extrapolated to time t_F.

2. The CSM state vector is extrapolated to time t_F.*

3. The LM state vector is initialized to the end-of-maneuver state vector.*

If a computation overflow occurs during any of the above W matrix extrapolations, a program alarm will result, the W Matrix control flag will be zeroed, and the extrapolation of the state vector will continue without the W matrix.

*These functions also done at the start of P68, and when P12 is halted after Average-G has started.

5.1.5.2 Altitude Parameter Convention

In the following programs and routines the display parameter of the vehicle altitude or trajectory pericenter or apocenter altitude is measured with respect to the earth launch pad radius magnitude, r_{LP}, when in earth orbit or the lunar landing site radius magnitude, r_{LS}, when in lunar orbit. The earth launch pad radius parameter, r_{LP}, is stored in fixed memory, and the lunar landing site radius, r_{LS}, is the magnitude of the landing site vector, $\underline{r}_{LS}$, stored in erasable memory.

P-21	Ground Track Determination Program	Sec. 5.6.5
P-30	External Delta V Program	Sec. 5.3.3.3.1
P-34, 74	Transfer Phase Initiation (TPI) Program	Sec. 5.4.2.4
R-30	Orbital Parameters Display Routine	Sec. 5.6.17

The above launch pad radius and landing site radius are also used in the following programs to compute pericenter altitudes prior to determining whether the pericenters are safe.

P-32, 72	Co-elliptic Sequence Initiation (CSI) Program	Sec. 5.4.2.2

The following programs are operated in lunar orbit only, and the displayed vehicle altitude or trajectory pericenter or apocenter altitude is referenced to the lunar landing site radius magnitude.

P-12	Powered Ascent Program	Sec. 5.3.5
P-63	Braking Phase Program	Sec. 5.3.4.6

P-64	Approach Phase Program	Sec. 5.3.4.6
P-66	Landing Phase Program	Sec. 5.3.4.6
P-70	DPS Abort Program	Sec. 5.4.3.1
P-71	APS Abort Program	Sec. 5.4.3.1

In the following programs a temporary r_{LS} can be displayed on the DSKY with respect to the original r_{LS} in erasable memory. The temporary altitude value is considered to be zero unless the astronaut keys in a change. In P52 the temporary value is used by the program but r_{LS} retains its original value in memory. In P57, when using alignment technique number 2, the temporary value replaces the old r_{LS} in memory.

P-52	IMU Realign Program	Sec. 5.6.2.1
P-57	Lunar Surface Align Program	Sec. 5.6.2.2

Program P68 also displays altitude, with a base of the original $\underline{r}_{LS}$ site (or if there has been a restart, with a base of the final state vector).

5.1.5.3 Lunar Landing Site Definition

A lunar landing site vector, $\underline{r}_{LS}$, in the Moon Fixed Coordinate System (Section 5.1.4.5) is stored in an LGC erasable memory location at all times. This landing site vector is normally stored in erasable memory prior to earth launch, and is either modified or verified prior to CSM-LM separation in lunar orbit by the CMC Orbital Navigation Program P-22 and astronaut LGC initialization procedures, or by the RTCC uplink program P-27. After the LGC initialization in lunar orbit prior to the descent orbit injection maneuver, the landing site vector $\underline{r}_{LS}$ is never changed until the Landing Confirmation Program P68 replaces it with a value based on the actual spacecraft position vector. The landing site vector, modified via N69 or during P64 by astronaut redesignations, occupies separate erasables, is in platform coordinates, and is initialized from $\underline{r}_{LS}$ at the beginning of P63. After lunar landing $\underline{r}_{LS}$ can be changed by P57, (the Lunar Surface Align Program), or by Program P-27. The lunar landing and lunar launch LM IMU alignments are both referenced to the landing site vector stored in the LGC at the time of alignment.

5.2 COASTING FLIGHT NAVIGATION

5.2.1 GENERAL COMMENTS

The LGC Coasting Flight Navigation Routines which are presented in Sections 5.2.2 through 5.2.5 are used during non-thrusting phases of the Apollo mission. The basic objective of the navigation routines is to maintain estimates of the position and velocity vectors of both the CSM and the LM. Let $\underline{r}$ and $\underline{v}$ be the estimates of a vehicle's position and velocity vectors, respectively. Then, the six-dimensional state vector, $\underline{x}$, of the spacecraft is defined by

$$\underline{x} = \begin{pmatrix} \underline{r} \\ \\ \underline{v} \end{pmatrix}$$

Coasting Flight Navigation is accomplished by extrapolating the state vector, $\underline{x}$, by means of the Coasting Integration Routine (Section 5.2.2), and updating or modifying this estimated state using Rendezvous Radar (RR) tracking data by the recursive method of navigation (Sections 5.2.3 - 5.2.5).

The Coasting Integration Routine (Section 5.2.2) is used by other navigation and targeting routines to extrapolate the following:

1) Present estimated LM state vector

2) Present estimated CSM state vector

3) An arbitrary specified state vector, such as the predicted result of a maneuver

State vector extrapolation is accomplished by means of Encke's method of differential accelerations. The motion of a spacecraft is dominated by the conic orbit which would result if the spacecraft were in a central force field. In Encke's method the differential equations for the deviations from conic motion are integrated numerically. This technique is in contrast to a numerical integration of the differential equations for the total motion, and it provides a more accurate orbit extrapolation. The numerical integration is accomplished by means of Nystrom's method which gives fourth-order accuracy while requiring only three computations of the derivatives per time step. The usual fourth-order Runge-Kutta integration methods require four derivative computations per time step.

Regardless of the accuracy of the state vector extrapolation, errors in the initial conditions will propagate and soon grow to intolerable size. Thus, it is necessary periodically to obtain additional data in the form of either new state vector estimates or modifications to the current state vector estimates. These state vector modifications are computed from navigation data obtained by means of navigation measurements.

The LM PGNCS uses RR tracking data to compute state vector changes. Navigation measurement data are used to update state vector estimates during rendezvous and lunar surface navigation procedures. These two navigation procedures will be used normally during all LM-CSM lunar-orbit rendezvous phases and the LM lunar surface prelaunch phase, respectively, in the lunar landing mission. However, in order to provide for alternate mission capability, the rendezvous navigation procedure can be used near the moon or the earth.

Although the state vector of the LM is six-dimensional, it is not necessary that the quantities estimated during a particular navigation procedure be the position and velocity vectors of the LM. A variety of "estimated state vectors", not necessarily of six-dimensions, are used.

In order to achieve desired rendezvous objectives, it is necessary to expand the rendezvous navigation procedure to nine dimensions, and to include in the estimation the constant RR angle biases. The estimated state vector that is used in rendezvous navigation is given by

$$\underline{x} = \begin{pmatrix} \underline{r} \\ \underline{v} \\ \underline{b}\text{ias} \end{pmatrix}$$

where $\underline{r}$ and $\underline{v}$ are the estimated position and velocity vectors of either the LM or the CSM, and $\underline{b}$ias is a vector whose components are the estimates of the RR angle biases. Normally the LM state vector is updated, but the astronaut can select the CSM update mode. The selection of the vehicle update mode is based primarily upon

which vehicle's state vector is most accurately known initially, and which vehicle is controlling the rendezvous maneuvers.

In order to estimate the RR angle biases, it is necessary to restrict the LM attitude during RR tracking. This attitude restriction involves controlling the LM +Z-axis to be within 30° of the tracking line-of-sight and is described in Section 5.2.4.1. During normal RR tracking and update of the navigation equations with RR data the LM +Z-axis is actually directed along the line-of-sight to the CSM on a continuous basis as described in Sections 5.2.4.2.1 and 5.2.4.4.

During the LM lunar surface prelaunch phase of the lunar landing mission RR tracking is used for navigation. In this mode, however, the previously mentioned LM attitude restriction obviously cannot be met, and only the RR range and range rate data are used. Also, since it is assumed that the landing site is well known, the estimated state vector that is used in lunar surface navigation is the standard six-dimensional CSM state vector.

Navigation data is incorporated into the state vector estimates by means of the Measurement Incorporation Routine (Section 5.2.3) which has both six- and nine-dimensional modes. The Measurement Incorporation Routine is a subroutine of the following navigation routines:

1) Rendezvous Navigation Routine (Section 5.2.4.2)

2) Lunar Surface Navigation Routine (Section 5.2.5.4)

Simplified functional diagrams of the navigation programs which use these routines are given in Figs. 2.1-1 and 2.1-2, respectively.

In rendezvous navigation, estimated LM and CSM position and velocity vectors are obtained at required times by means of the

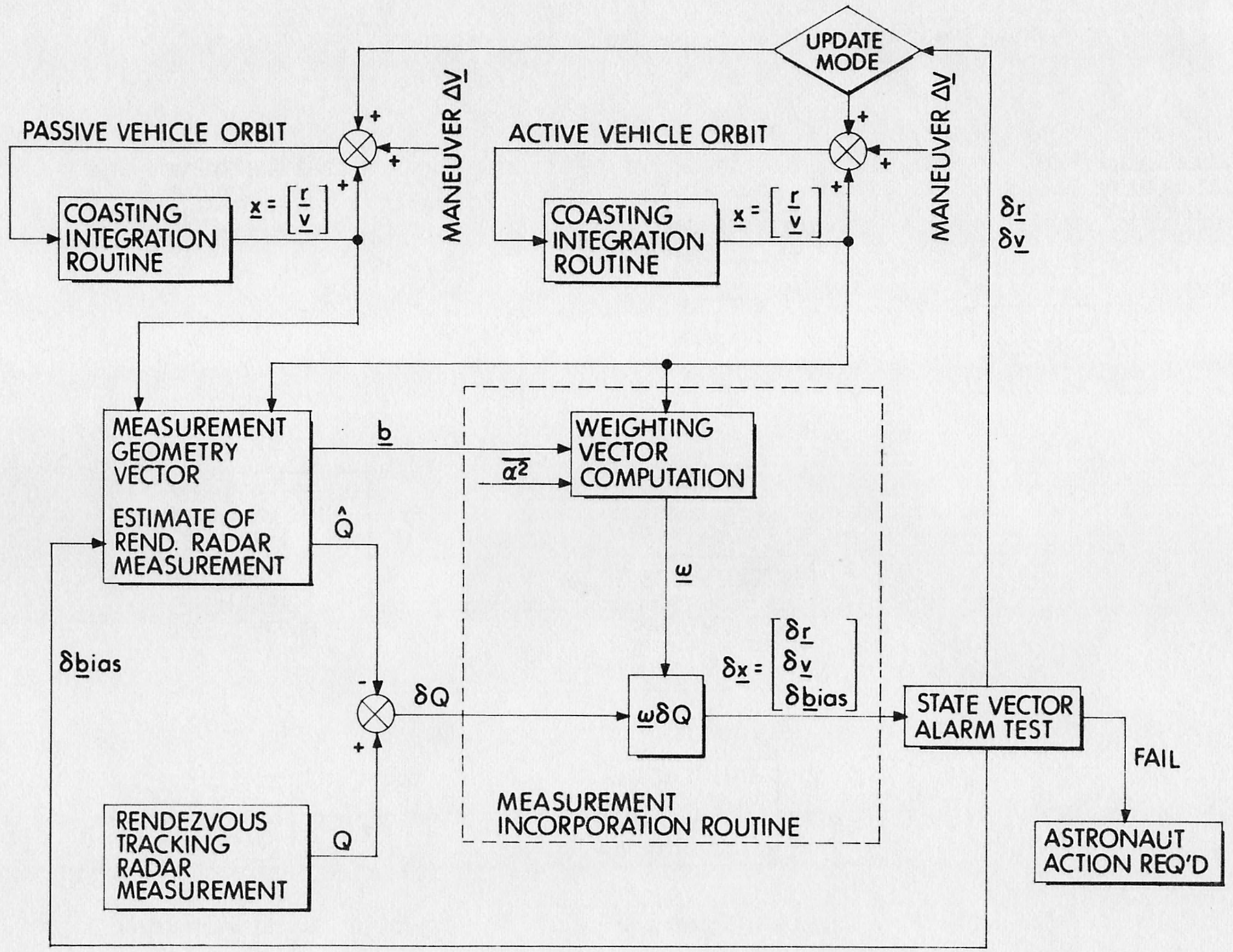

Figure 2.1-1 Simplified LGC Rendezvous Navigation Functional Diagram

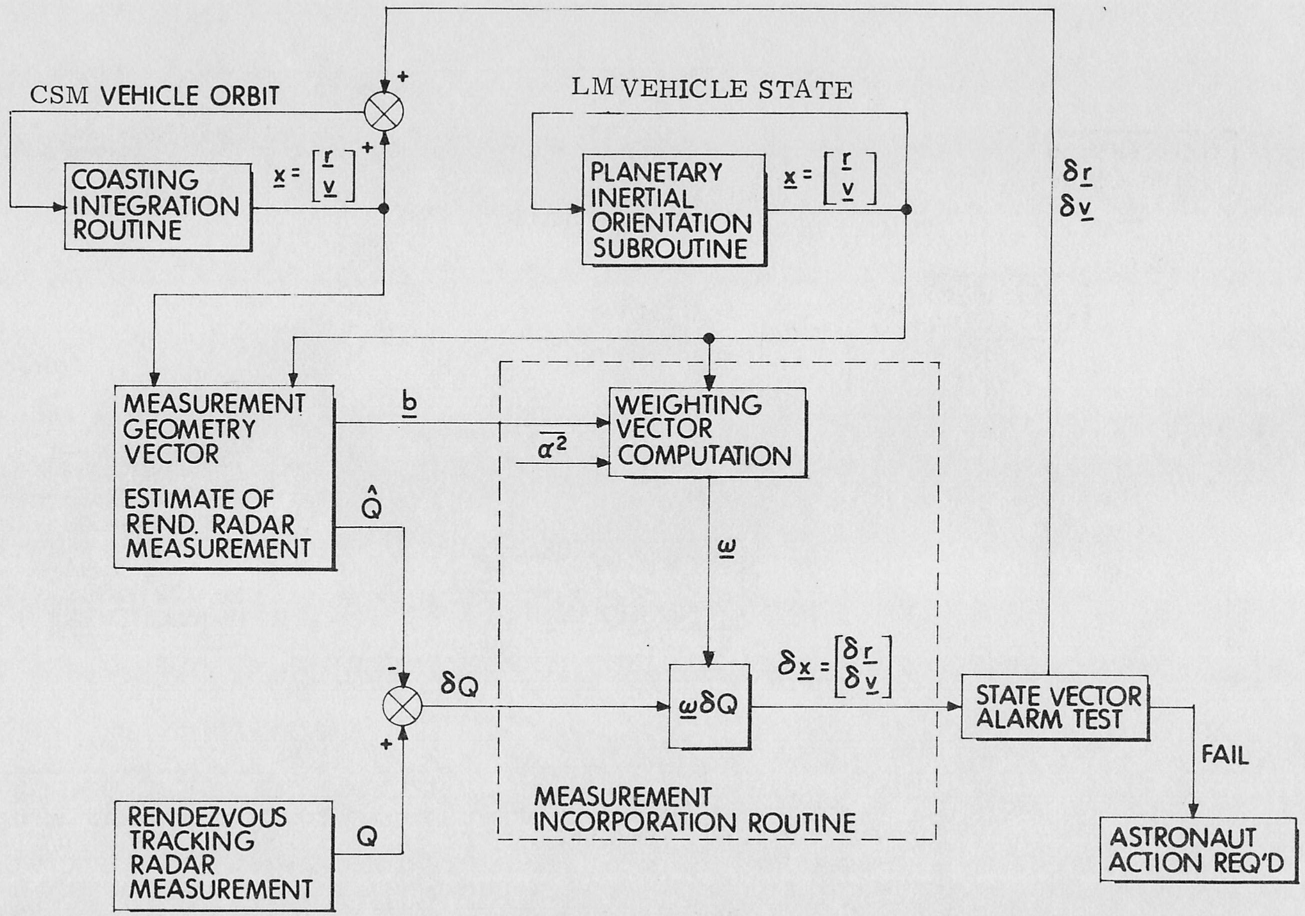

Figure 2.1-2 Simplified Lunar Surface Navigation Functional Diagram

Coasting Integration Routine (Section 5.2.2). The Measurement Incorporation Routine (Section 5.2.3) is used to incorporate the measurement data into the state vector estimates. In lunar surface navigation, the same process is performed except that the estimated LM state vector is obtained by means of the Planetary Inertial Orientation Subroutine (Section 5.5.2).

The navigation procedure, which is illustrated in simplified form in Figs. 2.1-1 and 2.1-2, involves computing an estimated tracking measurement, $\hat{Q}$, based on the current state vector estimates. This estimated measurement is then compared with the actual tracking measurement Q (RR tracking data in the LGC) to form a measured deviation δQ. A statistical weighting vector, $\underline{\omega}$, is computed from statistical knowledge of state vector uncertainties and tracking performance, $\overline{\alpha^2}$, plus a geometry vector, $\underline{b}$, determined by the type of measurement being made. The weighting vector, $\underline{\omega}$, is defined such that a statistically optimum linear estimate of the deviation, $\delta \underline{x}$, from the estimated state vector is obtained when the weighting vector is multiplied by the measured deviation δQ. The vectors $\underline{\omega}$, $\underline{b}$ and $\delta \underline{x}$ are of six or nine dimensions depending upon the dimension of the state vector being estimated.

In an attempt to prevent unacceptably large incorrect state vector changes, certain validity tests have been included in the LGC navigation routines.

In the Rendezvous and Lunar Surface Navigation Routines (Sections 5.2.4.2 and 5.2.5.4) measurement data is processed periodically (once or twice per minute), and it is desirable that the CSM be tracked during the entire rendezvous phase up to the manual terminal maneuver. If the magnitudes of the changes in the estimated position and velocity vectors, δr and δv, respectively, are both less than preset tracking alarm levels, then the selected vehicle's state vector is automatically updated by the computed deviation, $\delta \underline{x}$, and no special display is presented, except that the tracking

measurement counter is incremented by one. If either δr or δv exceeds its alarm level, then the state vector is not updated, and the astronaut is alerted to this condition by a special display of δr and δv.

In this case the astronaut should place the RR under manual control and make the necessary radar operating and side lobe checks to verify main lobe lock-on and tracking conditions. After the tracking has been verified, and navigation data has again been acquired, the astronaut has the option of commanding a state vector update if the tracking alarm is again exceeded, or of repeating further RR checks before incorporating the measurement data. If the astronaut cannot verify the tracking, then he can terminate the program and try to achieve tracking conditions at a later time.

The tracking alarm criterion is incorporated in the navigation routines to alert the astronaut to the fact that the state vector update is larger than normally expected, and to prevent the estimated state vector from automatically being updated in such cases. The update occurs only by specific command of the astronaut. The tracking alarm level beyond which updating is suspended is primarily chosen to avoid false acquisition and tracking conditions. There is a low probability that the alarm level will be exceeded in the LGC if the estimated state vectors are essentially correct, since the RR Designation and Data Read Routines have partial internal checks for side-lobe acquisition before tracking data are incorporated in the navigation routines. The most probable condition for the state vector update alarm level being exceeded in the LGC is, therefore, initial acquisition and tracking in the case where a poor estimate of either the CSM or LM state vector exists. In this case the astronaut would have to command the initial state vector update, after which

the tracking alarm level would seldom be exceeded during the remainder of the navigation phase. This statement is true only if the estimated state vector of the vehicle performing a rendezvous maneuver is updated. This is done by the Average-G Routine if the LM is the active vehicle and a powered flight program (P40, P41, P42, or P47) has been performed. Otherwise, the state vector is updated by a DSKY entry, using P76 if the CSM is the active vehicle (Section 5.6.16) or P77 if the LM is the active vehicle and a powered flight program has not been performed (Section 5.6.16).

The displayed values of δr and δv which have not passed the tracking alarm test will depend upon the statistical parameters stored in the LGC and upon the following types of errors:

Type 1: Errors in the current state vector estimates

Type 2: Errors in alignment of the IMU

Type 3: Reasonable RR tracking performance errors

Type 4: A PGNCS or RR failure resulting in false acquisition

The existence of Type 1 errors is precisely the reason that the RR tracking is being done. It is the function of the navigation to decrease Type 1 errors in the presence of noise in the form of errors of Types 2 and 3. Since the RR tracking should not be performed unless the IMU is well aligned and the PGNCS and RR are functioning properly, it follows that the purpose of the state vector change validity check is to discover a Type 4 error. As previously mentioned, there is a low probability of this type of error occurring.

Based upon the last time that the state vector was updated and when the IMU last was realigned, very crude reasonable values for δr and δv can be generated by the astronaut. The LGC will provide no information to assist the astronaut in his estimates of reasonable values for δr and δv.

The parameters required to initialize the navigation routines (Sections 5. 2. 4. 2 and 5. 2. 5. 4) are the initial estimated CSM state vector, plus the initial estimated LM state vector for the Rendezvous Navigation Routine or the estimated landing site for the Lunar Surface Navigation Routine, initial state vector estimation error covariance matrices in the form of prestored diagonal error transition matrices (as defined in Section 5. 2. 2. 4), and a priori measurement error variances. The basic input to the navigation routines is RR tracking data which is automatically acquired by the Data Read Routine. The primary results of the navigation routines are the estimated LM and CSM state vectors. The various guidance targeting modes outlined in Section 5. 4 are based on the state vector estimates which result from these navigation routines.

5.2.2 COASTING INTEGRATION ROUTINE *

5.2.2.1 General Comments

During all coasting phase navigation procedures, an extrapolation of position and velocity by numerical integration of the equations of motion is required. The basic equation may be written in the form

$$\frac{d^2}{dt^2}\underline{r}(t) + \frac{\mu_P}{r^3}{}^{**}\,\underline{r}(t) = \underline{a}_d(t) \qquad (2.2.1)$$

where μ_P is the gravitational constant of the primary body, and $\underline{a}_d(t)$ is the vector acceleration which prevents the motion of the vehicle (CSM or LM) from being precisely a conic with focus at the center of the primary body. The Coasting Integration Routine is a precision integration routine in which all significant perturbation effects are included. The form of the disturbing acceleration $\underline{a}_d(t)$ depends on the phase of the mission.

An approximate extrapolation of a vehicle state vector in which the disturbing acceleration, $\underline{a}_d(t)$ of Eq. (2.2.1), is set to zero may be accomplished by means of the Kepler subroutine (Section 5.5.5).

The LGC Coasting Integration Routine is restricted to earth or lunar orbit and is not to be used in cislunar-midcourse space. The routine does not contain the capability of computing the gravitational perturbations of the sun or the other body (moon or earth); and, therefore, cannot provide accurate midcourse integration.

5.2.2.2 Encke's Method

If $\underline{a}_d$ is small compared with the central force field, direct integration of Eq. (2.2.1) is inefficient. Therefore, the extrapolation will be accomplished using the technique of differential accelerations attributed to Encke.

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*This section does not reflect the entire LGC Coasting Integration Routine which is identical to that contained in the CMC. These two routines have been kept identical for the purpose of efficient flight program production. However, certain control constants are set in the LGC so that the midcourse perturbation calculations are locked out. Only those equations which can be executed are documented here.

**In the remainder of Section 5.2 the subscript P will denote primary body (earth or moon). When the body is known, then the subscripts E and M will be used for earth and moon, respectively. The vehicle will be indicated by the subscripts C for CSM and L for LM.

At time t_0 the position and velocity vectors, $\underline{r}_0$ and $\underline{v}_0$, define an osculating conic orbit. The position and velocity vectors in the conic orbit, $\underline{r}_{con}(t)$ and $\underline{v}_{con}(t)$, respectively, will deviate by a small amount from the actual position and velocity vectors.

The conic position and velocity at time t are computed as shown in Section 5.5.5. Required in this calculation is the variable x which is the root of Kepler's equation. In order to minimize the number of iterations required in solving Kepler's equation, an estimate of the correct solution for x is obtained as follows:

Let

$$\tau = t - t_0 \tag{2.2.2}$$

During the previous computation cycle the values

$$\underline{r}' = \underline{r}_{con}\left(\tau - \frac{\Delta t}{2}\right)$$

$$\underline{v}' = \underline{v}_{con}\left(\tau - \frac{\Delta t}{2}\right) \tag{2.2.3}$$

$$x' = x\left(\tau - \frac{\Delta t}{2}\right)$$

were computed. A trial value of $x(\tau)$ is obtained from

$$x_t = x' + s\left[1 - \gamma\, s\,(1 - 2\,\gamma\, s) - \frac{1}{6}\left(\frac{1}{r'} - \alpha\right) s^2\right] \tag{2.2.4}$$

where

$$s = \frac{\sqrt{\mu_P}}{r'}\left(\frac{\Delta t}{2}\right)$$

$$\gamma = \frac{\underline{r}' \cdot \underline{v}'}{2r'\sqrt{\mu_P}} \tag{2.2.5}$$

$$\alpha = \frac{2}{r'} - \frac{v'^2}{\mu_P}$$

After specification of $\underline{r}_0$, $\underline{v}_0$, x_t and τ, the Kepler subroutine (Section 5.5.5) is used to compute $\underline{r}_{con}(\tau)$, $\underline{v}_{con}(\tau)$, and $x(\tau)$.

The true position and velocity vectors will deviate from the conic position and velocity since $\underline{a}_d$ is not zero. Let

$$\begin{aligned} \underline{r}(t) &= \underline{\delta}(t) + \underline{r}_{con}(t) \\ \underline{v}(t) &= \underline{\nu}(t) + \underline{v}_{con}(t) \end{aligned} \tag{2.2.6}$$

where $\underline{\delta}(t)$ and $\underline{\nu}(t)$ are the position and velocity deviations from the conic. The deviation vector $\underline{\delta}(t)$ satisfies the differential equation

$$\frac{d^2}{dt^2}\,\underline{\delta}(t) = -\frac{\mu_P}{r_{con}^3(t)}\left[f(q)\;\underline{r}(t) + \underline{\delta}(t)\right] + \underline{a}_d(t) \tag{2.2.7}$$

subject to the initial conditions

$$\underline{\delta}(t_0) = \underline{0}, \quad \underline{\nu}(t_0) = \underline{0} \tag{2.2.8}$$

where

$$q = \frac{(\underline{\delta} - 2\underline{r}) \cdot \underline{\delta}}{r^2} \tag{2.2.9}$$

$$f(q) = q \frac{3 + 3q + q^2}{1 + (1 + q)^{3/2}} \qquad (2.2.10)$$

The first term on the right-hand side of Eq. (2.2.7) must remain small, i.e., of the same order as $\underline{a}_d(t)$, if the method is to be efficient. As the deviation vector $\underline{\delta}(t)$ grows in magnitude, this term will eventually increase in size. Therefore, in order to maintain the efficiency of the method, a new osculating conic orbit should be defined by the total position and velocity vectors $\underline{r}(t)$ and $\underline{v}(t)$. The process of selecting a new conic orbit from which to calculate deviations is called rectification. When rectification occurs, the initial conditions for the differential equation for $\underline{\delta}(t)$, as well as the variables τ and x, are again zero.

5.2.2.3 Disturbing Acceleration

The form of the disturbing acceleration $\underline{a}_d(t)$ that is used in Eq. (2.2.1) depends on the phase of the mission. In earth or lunar orbit, only the gravitational perturbations arising from the non-spherical shape of the primary body need be considered. Let $\underline{a}_{dP}$ be the acceleration due to the non-spherical gravitational perturbations of the primary body. Then, for the earth

$$\underline{a}_{dE} = \frac{\mu_E}{r^2} \sum_{i=2}^{4} J_{iE} \left(\frac{r_E}{r}\right)^i \left[P'_{i+1} (\cos\phi)\, \underline{u}_r - P'_i (\cos\phi)\, \underline{u}_z\right] \qquad (2.2.11)$$

where

$$P_2'(\cos\phi) = 3\cos\phi$$

$$P_3'(\cos\phi) = \frac{1}{2}(15\cos^2\phi - 3)$$

$$P_4'(\cos\phi) = \frac{1}{3}(7\cos\phi\, P_3' - 4P_2') \qquad (2.2.12)$$

$$P_5'(\cos\phi) = \frac{1}{4}(9\cos\phi\, P_4' - 5P_3')$$

are the derivatives of Legendre polynominals,

$$\cos\phi = \underline{u}_r \cdot \underline{u}_z$$

$$\underline{u}_z = \begin{pmatrix} 0 \\ 0 \\ 1 \end{pmatrix} \qquad (2.2.13)$$

and J_2, J_3, J_4 are the coefficients of the second, third, and fourth harmonics of the earth's potential function. The vectors $\underline{u}_r$ and $\underline{u}_z$ are unit vectors in the direction of $\underline{r}$ and the polar axis of the earth, respectively, and r_E is the equatorial radius of the earth.

In the case of the moon,

$$\underline{a}_{dM} = \frac{\mu_M}{r^2}\Bigg\{\sum_{i=2}^{4} J_{iM}\left(\frac{r_M}{r}\right)^i \left[P_{i+1}'(\cos\phi)\,\underline{u}_r - P_i'(\cos\phi)\,\underline{u}_z\right]$$

$$+ 3J_{22}\left(\frac{r_M}{r}\right)^2 \left[\frac{-5\left(x_M^2 - y_M^2\right)}{r^2}\,\underline{u}_r + \frac{2x_M}{r}\,\underline{u}_x - \frac{2y_M}{r}\,\underline{u}_y\right]$$

$$+ \frac{3}{2}C_{31}\left(\frac{r_M}{r}\right)^3 \Bigg[\frac{5x_M}{r}(1 - 7\cos^2\phi)\,\underline{u}_r + (5\cos^2\phi - 1)\,\underline{u}_x$$

$$+ \frac{10x_M z_M}{r^2}\,\underline{u}_z\Bigg]\Bigg\} \qquad (2.2.14)$$

where:

$\underline{u}_r$ is the unit position vector in reference coordinates;

$\underline{u}_x$ is planetary $\begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}$ transformed to reference coordinates;

$\underline{u}_y$ is planetary $\begin{pmatrix} 0 \\ 1 \\ 0 \end{pmatrix}$ transformed to reference coordinates;

$\underline{u}_z$ is planetary $\begin{pmatrix} 0 \\ 0 \\ 1 \end{pmatrix}$ transformed to reference coordinates;

and x_M, y_M, z_M are the components of $\underline{r}$ in planetary coordinates, which are computed by use of the Planetary Inertial Orientation Subroutine (Section 5.5.2). In addition, r_M is the mean lunar radius; and J_{22}, C_{31} are the coefficients of the terms describing the asymmetry about the pole of the moon's gravity; and the remaining symbols are defined as in Eq. (2.2.11).

5.2.2.4 Error Transition Matrix

The position and velocity vectors as maintained in the computer are only estimates of the true values. As part of the navigation technique it is necessary also to maintain statistical data in the computer to aid in the processing of navigation measurements.

If $\underline{\epsilon}(t)$ and $\underline{\eta}(t)$ are the errors in the estimates of the position and velocity vectors, respectively, then the six-dimensional correlation matrix E(t) is defined by

$$E_6(t) = \begin{pmatrix} \overline{\underline{\epsilon}(t)\ \underline{\epsilon}(t)^T} & \overline{\underline{\epsilon}(t)\ \underline{\eta}(t)^T} \\ \\ \overline{\underline{\eta}(t)\ \underline{\epsilon}(t)^T} & \overline{\underline{\eta}(t)\ \underline{\eta}(t)^T} \end{pmatrix} \tag{2.2.15}$$

In certain applications it becomes necessary to expand the state vector and the correlation matrix to more than six dimensions so as to include estimation of landmark locations in the CMC during orbit navigation, and rendezvous radar tracking biases in the LGC during the rendezvous navigation procedure. For this purpose a nine-dimensional correlation matrix is defined as follows :

$$E(t) = \begin{pmatrix} \begin{matrix} & E_6(t) & \end{matrix} & \begin{matrix} \overline{\underline{\epsilon}(t)\ \underline{\beta}^T} \\ \overline{\underline{\eta}(t)\ \underline{\beta}^T} \end{matrix} \\ \begin{matrix} \overline{\underline{\beta}\ \underline{\epsilon}(t)^T} & \overline{\underline{\beta}\ \underline{\eta}(t)^T} \end{matrix} & \overline{\underline{\beta}\ \underline{\beta}^T} \end{pmatrix} \tag{2.2.16}$$

where the components of the three-dimensional vector $\underline{\beta}$ are the errors in the estimates of three variables which are estimated in addition to the components of the spacecraft state vector.

In order to take full advantage of the operations provided by the interpreter in the computer, the correlation matrix will be restricted to either six or nine dimensions. If, in some navigation procedure, only one or two additional items are to be estimated, then a sufficient number of dummy variables will be added to the desired seven- or eight-dimensional state vector to make it nine-dimensional.

Rather than use the correlation matrix in the navigation procedure, it is more convenient to utilize a matrix W(t), called the error transition matrix, and defined by

$$E(t) = W(t)\, W(t)^T \tag{2.2.17}$$

Extrapolation of the nine-dimensional matrix W(t) is made by direct numerical integration of the differential equation

$$\frac{d}{dt} W(t) = \begin{pmatrix} O & I & O \\ G(t) & O & O \\ O & O & O \end{pmatrix} W(t) \tag{2.2.18}$$

where G(t) is the three-dimensional gravity gradient matrix, and I and O are the three-dimensional identity and zero matrices, respectively. If the W matrix is partitioned as

$$W = \begin{pmatrix} \underline{w}_0 & \underline{w}_1 & \cdots & \underline{w}_8 \\ \underline{w}_9 & \underline{w}_{10} & \cdots & \underline{w}_{17} \\ \underline{w}_{18} & \underline{w}_{19} & \cdots & \underline{w}_{26} \end{pmatrix} \tag{2.2.19}$$

then,

$$\left.\begin{aligned} &\frac{d}{dt}\,\underline{w}_i(t) = \underline{w}_{i+9}(t) \\ &\frac{d}{dt}\,\underline{w}_{i+9}(t) = G(t)\,\underline{w}_i(t) \\ &\frac{d}{dt}\,\underline{w}_{i+18}(t) = \underline{0} \end{aligned}\right\} \quad i = 0,\ 1, \ldots, 8 \tag{2.2.20}$$

The extrapolation may be accomplished by successively integrating the vector differential equations

$$\frac{d^2}{dt^2}\,\underline{w}_i(t) = G(t)\,\underline{w}_i(t) \qquad i = 0, 1, \ldots, 8 \tag{2.2.21}$$

The gravity gradient matrix G(t) for earth or lunar orbit is given by

$$G(t) = \frac{\mu_P}{r^5(t)}\left[3\,\underline{r}(t)\,\underline{r}(t)^T - r^2(t)\,I\right] \tag{2.2.22}$$

Thus, if D is the dimension of the matrix W(t) for the given navigation procedure, the differential equations for the $\underline{w}_i(t)$ vectors are

$$\frac{d^2}{dt^2}\underline{w}_i(t) = \frac{\mu_P}{r^3(t)}\left\{3\left[\underline{u}_r(t)\cdot\underline{w}_i(t)\right]\underline{u}_r(t) - \underline{w}_i(t)\right\} \qquad (2.2.23)$$

$$(i = 0, 1, \ldots, D-1)$$

where $\underline{u}_r(t)$ is a unit vector in the direction of $\underline{r}(t)$.

It is possible for a computation overflow to occur during the W matrix integration if any element of the matrix exceeds its maximum value. This event is extremely unlikely because of the large scale factors chosen. The overflow occurs if

1) any element of the position part (upper third) of the W matrix becomes equal to or greater than 2^{19}m,

or

2) any element of the velocity part (middle third) of the W matrix becomes equal to or greater than one m/csec.

In addition, each element of the bias part (lower third) of the matrix must remain less than 2^{-5} radian, but this part does not change during integration.

If overflow should occur, an alarm results, the W matrix control flag is reset, and either new state vector estimates must be obtained from RTCC, or a sufficient number of navigation measurements must be made before the state vectors are used in any targeting or maneuver programs.

5.2.2.5 Numerical Integration Method

The extrapolation of navigational data requires the solution of a number of second-order vector differential equations, specifically Eqs. (2.2.7) and (2.2.23). These are all special cases of the form

$$\frac{d^2}{dt^2}\underline{y} = \underline{f}(\underline{y}, t) \tag{2.2.24}$$

Nystrom's method is particularly well suited to this form and gives an integration method of fourth-order accuracy. The second-order system is written

$$\begin{aligned} \frac{d}{dt}\underline{y} &= \underline{z} \\ \frac{d}{dt}\underline{z} &= \underline{f}(\underline{y}, t) \end{aligned} \tag{2.2.25}$$

and the formulas are summarized below.

$$\begin{aligned}
\underline{y}_{n+1} &= \underline{y}_n + \underline{\phi}(\underline{y}_n)\,\Delta t \\
\underline{z}_{n+1} &= \underline{z}_n + \underline{\psi}(\underline{y}_n)\,\Delta t \\
\underline{\phi}(\underline{y}_n) &= \underline{z}_n + \frac{1}{6}(\underline{k}_1 + 2\underline{k}_2)\,\Delta t \\
\underline{\psi}(\underline{y}_n) &= \frac{1}{6}(\underline{k}_1 + 4\underline{k}_2 + \underline{k}_3) \\
\underline{k}_1 &= \underline{f}(\underline{y}_n, t_n) \\
\underline{k}_2 &= \underline{f}\left(\underline{y}_n + \frac{1}{2}\underline{z}_n\,\Delta t + \frac{1}{8}\underline{k}_1(\Delta t)^2,\quad t_n + \frac{1}{2}\Delta t\right) \\
\underline{k}_3 &= \underline{f}\left(\underline{y}_n + \underline{z}_n\,\Delta t + \frac{1}{2}\underline{k}_2(\Delta t)^2,\ t_n + \Delta t\right)
\end{aligned} \tag{2.2.26}$$

For efficient use of computer storage as well as computing time the computations are performed in the following order:

1) Equation (2.2.7) is solved using the Nystrom formulas, Eqs.(2.2.26). It is necessary to preserve the values of the vector $\underline{r}$ at times t_n, $t_n + \Delta t/2$, $t_n + \Delta t$ for use in the solution of Eqs.(2.2.23).

2) Equations (2.2.23) are solved one-at-a-time using Eqs. (2.2.26) together with the values of $\underline{r}$ which resulted from the first step.

The variable Δt is the integration time step and should not be confused with τ, the time since rectification. The maximum value for Δt which can be used for precision integration, Δt_{max}, is computed from

$$\Delta t_{max} = \text{minimum} \left(0.3 \, \frac{r_{con}^{3/2}}{\sqrt{\mu_P}} \, , 4000 \text{ sec} \right) \qquad (2.2.27)$$

5.2 2.6 Coasting Integration Logic

Estimates of the state vectors of two vehicles (CSM and LM) will be maintained in the computer. In various phases of the mission it will be required to extrapolate a state vector either alone or with an associated W matrix of dimension six or nine.

To accomplish all of these possible procedures, as well as to solve the computer restart problem, three state vectors will be maintained in the computer. Let $\underline{x}_C$ and $\underline{x}_L$ be the estimated CSM and LM state vectors, respectively, and let $\underline{x}$ be a temporary state vector. The state vector $\underline{x}$ is a symbolic representation of the following set of variables:

$$
\begin{aligned}
\underline{r}_0 &= \text{rectification position vector} \\
\underline{v}_0 &= \text{rectification velocity vector} \\
\underline{r}_{con} &= \text{conic position vector} \\
\underline{v}_{con} &= \text{conic velocity vector} \\
\underline{\delta} &= \text{position deviation vector} \\
\underline{\nu} &= \text{velocity deviation vector} \\
t &= \text{time associated with } \underline{r}_{con}, \underline{v}_{con}, \underline{\delta} \text{ and } \underline{\nu} \\
\tau &= \text{time since rectification} \\
x &= \text{root of Kepler's equation} \\
P &= \text{primary body} = \begin{cases} 0 \text{ for earth} \\ 1 \text{ for moon} \end{cases}
\end{aligned}
\tag{2.2.28}
$$

The state vectors $\underline{x}_C$ and $\underline{x}_L$ represent an analogous set of variables.

The Coasting Integration Routine is controlled by the calling program by means of the two indicators D and V. The variable D indicates the dimension of the W matrix with

$$D = 0 \tag{2.2.29}$$

denoting that the state vector only is to be extrapolated. The variable V indicates the appropriate vehicle as follows:

$$V = \begin{cases} 1 \text{ for CSM} \\ 0 \text{ for LM} \\ -1 \text{ for state vector specified by calling program} \end{cases} \qquad (2.2.30)$$

In addition, the calling program must set the desired final time t_F; and, for V equal to -1, the desired state vector $\underline{x}$.

A simplified functional diagram of the Coasting Integration Routine is shown in Fig 2.2-1. In the figure the indicated state vector is being integrated to time t_F. The value of Δt for each time step is Δt_{max} (Eq. (2.2.27)) or the total time-to-go whichever is smaller. The integration is terminated when the computed value of Δt is less than ϵ_t.

Figure 2.2-2 illustrates in more detail the logic flow of this routine. In this figure the following items, which have not been discussed fully in the text, are explicitly illustrated:

1) Saving of $\underline{r}$ values for W matrix integration
2) Rectification procedure
3) Selection of disturbing acceleration

The logic flow shown in Fig. 2.2-2 is controlled by the switch F which is used to distinguish between state vector integration (F = 1) and W matrix integration (F = 0).

If the Coasting Integration Routine is requested to extrapolate the estimated LM state vector and the LM is on the lunar surface, then the routine will use the Planetary Inertial Orientation Subroutine (Section 5.5.2) to compute the desired LM position and velocity and the normal integration will not be performed. This procedure is not indicated in the figure.

There is a procedure for the emergency termination of the Coasting Integration Routine in order to permit correction of wrong erasable memory parameters. This emergency function is described in Section 5.6.11.

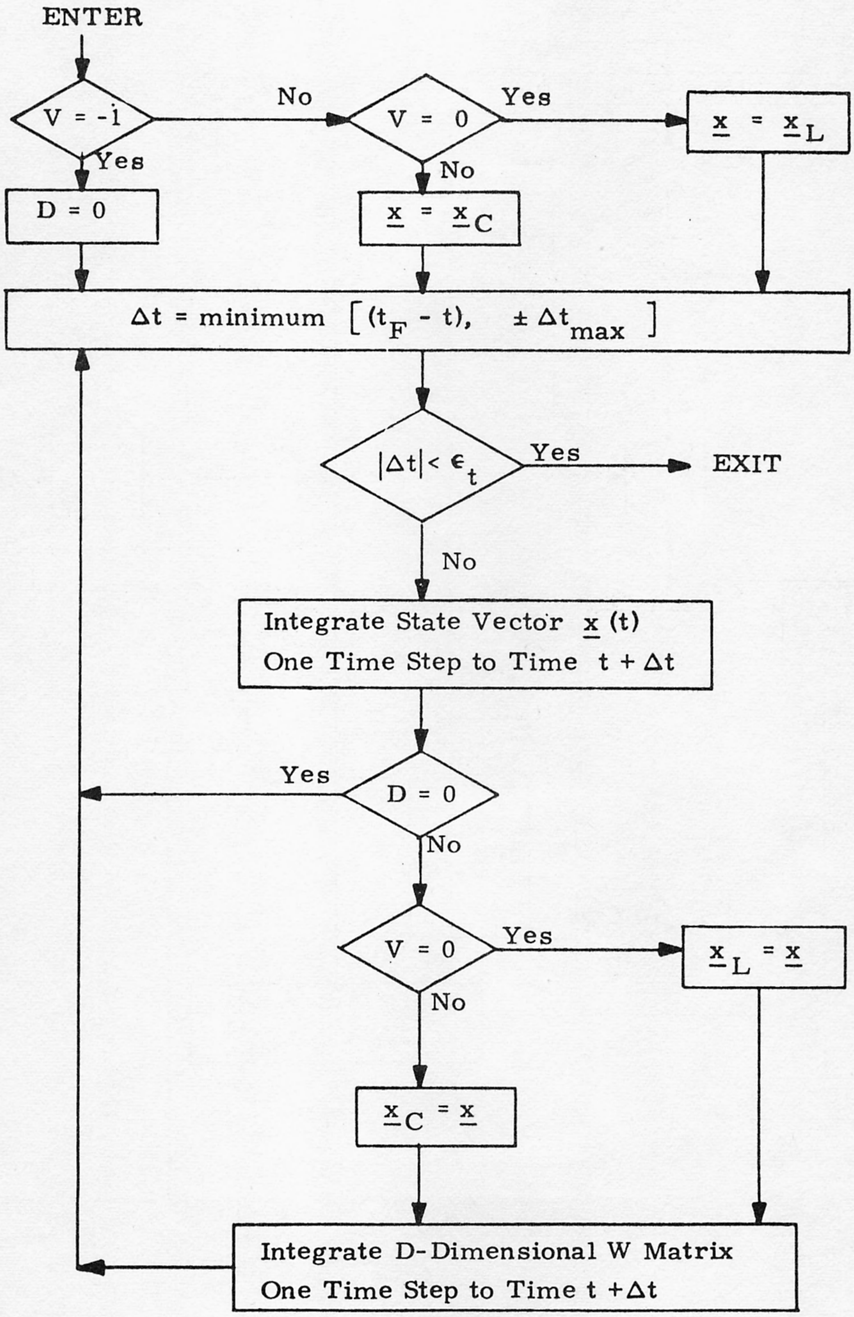


Figure 2.2-1 Simplified Coasting Integration Routine Logic Diagram

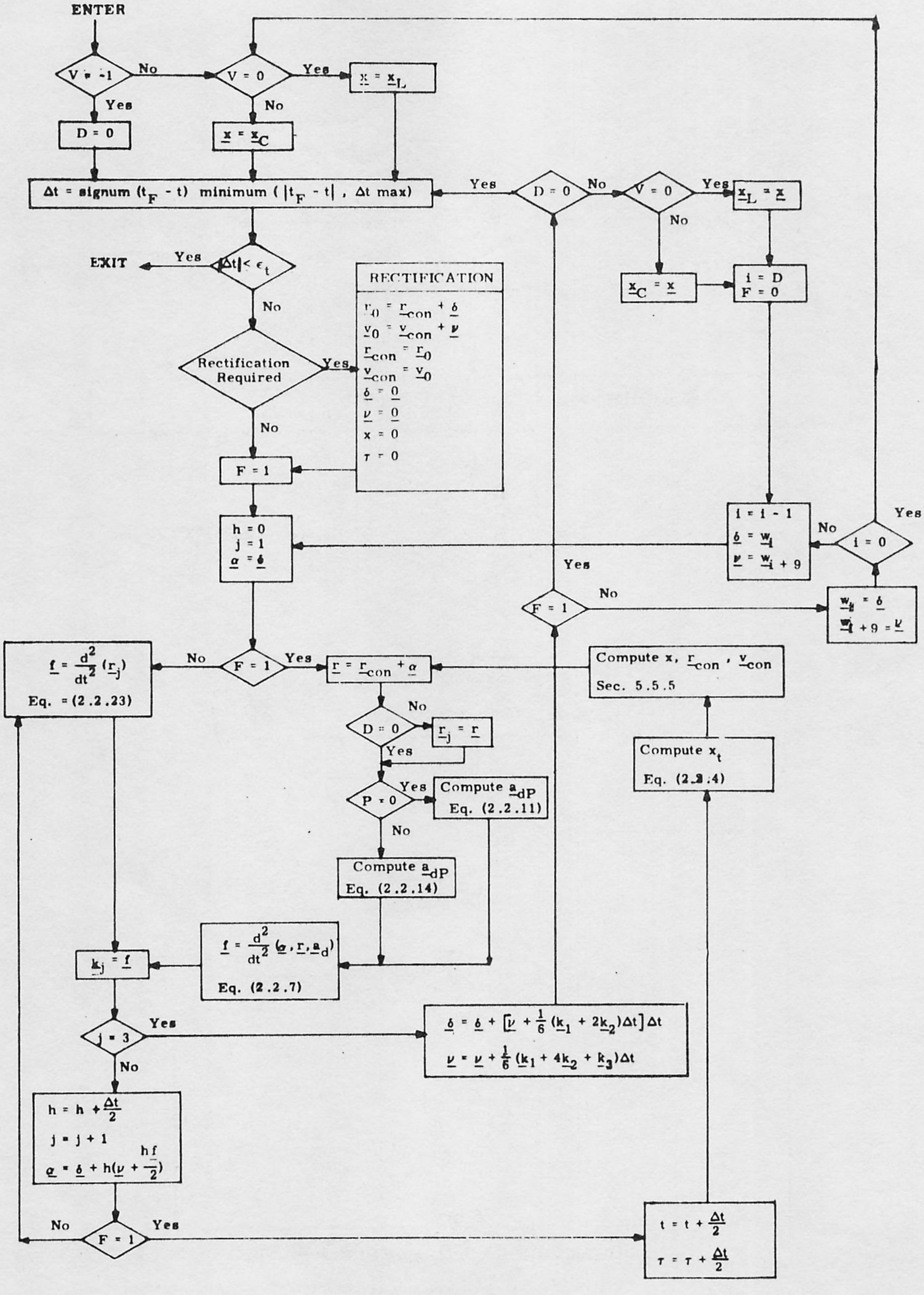


Figure 2. 2-2 Coasting Integration Routine Logic Diagram

In addition to the general criterion discussed in Section 5.2.2.2, the requirements for rectification (which are not shown in Fig. 2.2-2) are functions of

1) the computer word length,

2) the fact that the computations are performed in fixed-point arithmetic,

3) the scale factors of the variables, and

4) the accuracy of the Kepler Subroutine (Section 5.5.5).

If

$$\frac{\delta}{r_{con}} > 0.01$$

or if

$$\delta > \begin{cases} 0.75 \times 2^{22} \text{m for } P = 0 \\ 0.75 \times 2^{18} \text{m for } P = 1 \end{cases}$$

or if

$$\nu > \begin{cases} 0.75 \times 2^{3} \text{ m/csec for } P = 0 \\ 0.75 \times 2^{-1} \text{m/csec for } P = 1 \end{cases}$$

then rectification occurs at the point indicated in Fig. 2.2-2. Also, if the calculation of the acceleration (Eq. (2.2.7)) results in overflow (i.e. any component is equal to or greater than 2^{-16} m/csec^2 for P = 0, or 2^{-20} m/csec^2 for P = 1), then the program is recycled to the beginning of the time step and rectification is performed, provided that δ is not identically zero (which may occur if an attempt is made to extrapolate a state vector below the surface). In this exceptional case, an abort occurs with alarm code 20430_8.

5.2.3 MEASUREMENT INCORPORATION ROUTINE

Periodically it is necessary to update the estimated position and velocity vectors of the vehicle (CSM or LM) by means of navigation measurements. At the time a measurement is made, the best estimate of the state vector of the spacecraft is the extrapolated estimate denoted by $\underline{x}'$. The first six components of $\underline{x}'$ are the components of the estimated position and velocity vectors. In certain situations it becomes necessary to estimate more than six quantities. Then, the state vector will be of nine dimensions. From this state vector estimate it is possible to determine an estimate of the quantity measured. When the predicted value of this measurement is compared with the actual measured quantity, the difference is used to update the indicated state vector as well as its associated error transition matrix as described in Section 5.2.1. The error transition matrix, W, is defined in Section 5.2.2.4.

This routine is used to compute deviations to be added to the components of the estimated state vector, and to update the estimated state vector by these deviations provided the deviations pass a state vector update validity test as described in Section 5.2.1.

Let D be the dimension (six or nine) of the estimated state vector. Associated with each measurement are the following parameters which are to be specified by the program calling this routine:

$\underline{b}$ = Geometry vector of D dimensions

$\overline{\alpha^2}$ = A priori measurement error variance

δQ = Measured deviation, the difference between the quantity actually measured and the expected value based on the original value of the estimated state vector $\underline{x}'$.

The procedure for incorporating a measurement into the estimated state vector is as follows:

(1) Compute a D-dimensional $\underline{z}$ vector from

$$\underline{z} = W'^{T} \underline{b} \qquad (2.3.1)$$

where W' is the error transition matrix associated with $\underline{x}'$.

(2) Compute the D-dimensional weighting vector, $\underline{\omega}$, from

$$\underline{\omega}^{T} = \frac{1}{z^2 + \overline{\alpha^2}} \underline{z}^{T} W'^{T} \qquad (2.3.2)$$

(3) Compute the state vector deviation estimates from

$$\delta \underline{x} = \underline{\omega} \, \delta Q \qquad (2.3.3)$$

(4) If the data pass the validity test, update the state vector and the W matrix by

$$\underline{x} = \underline{x}' + \delta \underline{x} \qquad (2.3.4)$$

$$W = W' - \frac{\underline{\omega} \, \underline{z}^{T}}{1 + \sqrt{\dfrac{\overline{\alpha^2}}{z^2 + \overline{\alpha^2}}}} \qquad (2.3.5)$$

In order to take full advantage of the three-dimensional vector and matrix operations provided by the interpreter in the computer, the nine-dimensional W matrix will be stored sequentially in the computer as follows :

$$\underline{w}_0, \underline{w}_1, \ldots, \underline{w}_{26}$$

Refer to Section 5.2.2.4 for the definition of the W matrix. Define the three-dimensional matrices

$$W_0 = \begin{pmatrix} \underline{w}_0^T \\ \underline{w}_1^T \\ \underline{w}_2^T \end{pmatrix} \qquad W_1 = \begin{pmatrix} \underline{w}_3^T \\ \underline{w}_4^T \\ \underline{w}_5^T \end{pmatrix} \qquad \ldots \qquad W_8 = \begin{pmatrix} \underline{w}_{24}^T \\ \underline{w}_{25}^T \\ \underline{w}_{26}^T \end{pmatrix} \tag{2.3.6}$$

so that

$$W = \begin{pmatrix} W_0^T & W_1^T & W_2^T \\ W_3^T & W_4^T & W_5^T \\ W_6^T & W_7^T & W_8^T \end{pmatrix} \tag{2.3.7}$$

Let the nine-dimensional vectors $\delta\underline{x}$, $\underline{b}$, $\underline{\omega}$, and $\underline{z}$ be partitioned as follows:

$$\delta\underline{x} = \begin{pmatrix} \delta\underline{x}_0 \\ \delta\underline{x}_1 \\ \delta\underline{x}_2 \end{pmatrix} \qquad \underline{b} = \begin{pmatrix} \underline{b}_0 \\ \underline{b}_1 \\ \underline{b}_2 \end{pmatrix} \qquad \underline{\omega} = \begin{pmatrix} \underline{\omega}_0 \\ \underline{\omega}_1 \\ \underline{\omega}_2 \end{pmatrix} \qquad \underline{z} = \begin{pmatrix} z_0 \\ z_1 \\ \vdots \\ z_8 \end{pmatrix} = \begin{pmatrix} \underline{z}_0 \\ \underline{z}_1 \\ \underline{z}_2 \end{pmatrix}$$

(2.3.8)

Then, the computations shown in Eqs. (2. 3. 1) through (2. 3. 3) are performed as follows, using three-dimensional operations:

$$\underline{z}_i = \sum_{j=0}^{\frac{D}{3}-1} W'_{i+3j}\, \underline{b}_j$$

$$a = \sum_{j=0}^{\frac{D}{3}-1} \underline{z}_j \cdot \underline{z}_j + \overline{\alpha^2}$$

$$\underline{\omega}_i^T = \frac{1}{a} \sum_{j=0}^{\frac{D}{3}-1} \underline{z}_j^T\, W'_{3i+j} \tag{2.3.9}$$

$$\delta \underline{x}_i = \delta Q\, \underline{\omega}_i \qquad \left(i = 0, 1, \ldots, \frac{D}{3} - 1\right)$$

Equation (2. 3. 5) is written

$$\gamma = \frac{1}{1 + \sqrt{\alpha^2 / a}}$$

$$\underline{w}_{i+9j} = \underline{w}'_{i+9j} - \gamma\, z_i\, \underline{\omega}_j \qquad \begin{pmatrix} i = 0, 1, \ldots, D - 1 \\ j = 0, 1, \ldots, \frac{D}{3} - 1 \end{pmatrix} \tag{2.3.10}$$

The Measurement Incorporation Routine is divided into two subroutines, INCORP1 and INCORP2. The subroutine INCORP1 consists of Eqs. (2.3.9), while INCORP2 is composed of Eqs. (2.3.4) and (2.3.10). The method of using these subroutines is illustrated in Fig. 2.3-1.

Since the estimated position and velocity vectors are maintained in two pieces, conic and deviation from the conic, Eq. (2.3.4) cannot be applied directly. The estimated position and velocity deviations resulting from the measurement, $\delta\underline{x}_0$ and $\delta\underline{x}_1$, are added to the vectors $\underline{\delta}$ and $\underline{\nu}$, the position and velocity deviations from the conics, respectively. Since $\underline{\delta}$ and $\underline{\nu}$ are maintained to much higher accuracy than the conic position and velocity vectors, a possible computation overflow situation exists whenever Eq. (2.3.4) is applied. If overflow does occur, then it is necessary to reinitialize the Coasting Integration Routine (Section 5.2.2) by the process of rectification as described in Section 5.2.2.2. The logic flow of the subroutine INCORP2 is illustrated in detail in Fig. 2.3-2.

Overflow occurs when

$$\delta \geq \begin{cases} 2^{22} \text{ m for } P = 0 \\ 2^{18} \text{ m for } P = 1 \end{cases}$$

or

$$\nu \geq \begin{cases} 2^{3} \text{ m/csec for } P = 0 \\ 2^{-1} \text{ m/csec for } P = 1 \end{cases}$$

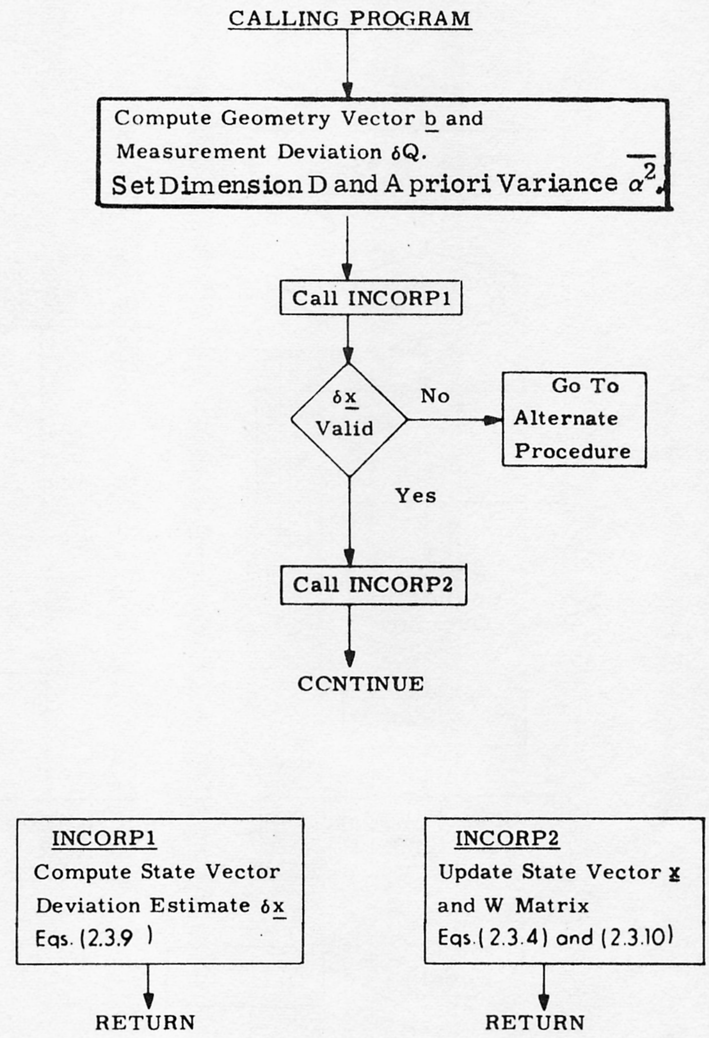


Fig. 2.3-1 Measurement Incorporation Procedure

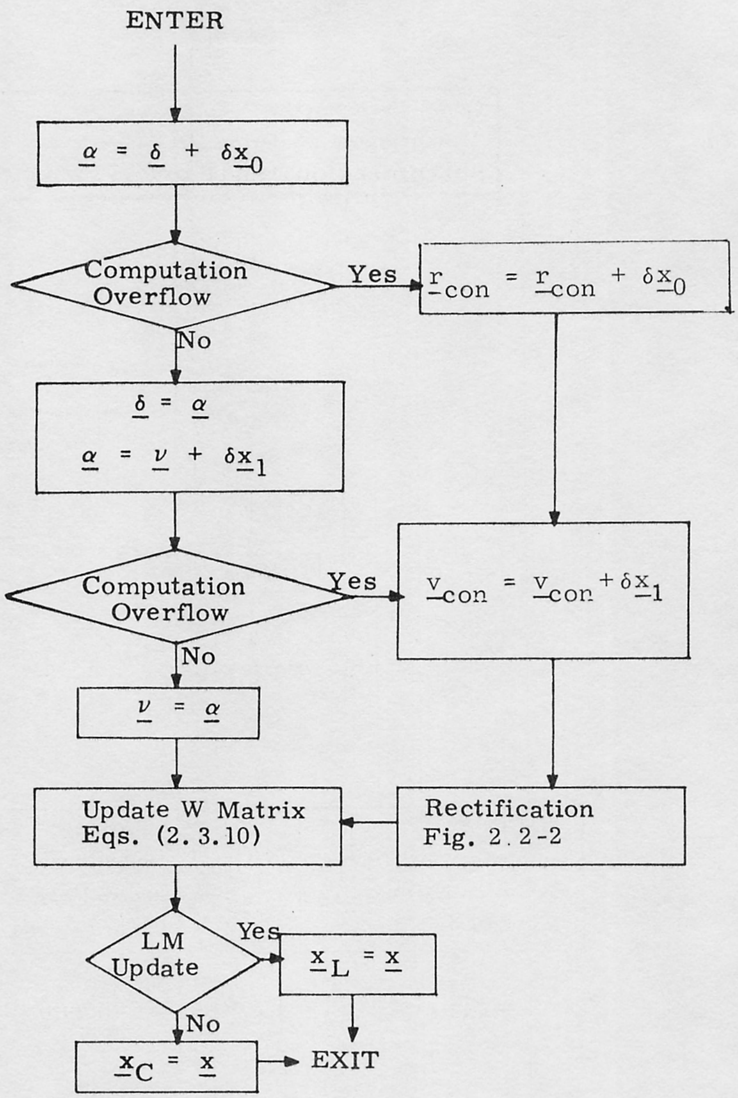


Fig. 2.3-2 INCORP2 Subroutine Logic Diagram

5.2.4 RENDEZVOUS NAVIGATION PROGRAM

5.2.4.1 Target Acquisition Routine

One of the first functions performed by the Rendezvous Navigation Program is to use the Target Acquisition Routine (Fig. 2.4-1) to establish lock-on of the LM Rendezvous Radar (RR) with the transponder on the CSM. Since the problem of acquiring the CSM with the RR is essentially the same for both the Rendezvous and Lunar Surface Navigation Programs (P-20 and P-22, respectively), the same Acquisition Routine is used for both except for certain differences in operation. For the moment, however, most of the operational details presented are for the case when the Rendezvous Navigation Program is being used. In either case it is assumed at the start of the routine that the RR is on and has been permitted to warm-up to operating conditions.

There are three modes (RR LGC, RR Manual, and RR Search) for controlling the RR in target acquisition. The RR LGC and RR Manual Modes can be selected by the astronaut at the beginning of program P-20 by a procedure described later. However, the RR Search Mode can be selected only after the RR LGC Mode has failed to acquire the target. The RR Manual Mode is not used in P-22.

Prior to using the Target Acquisition Routine, program P-20 sets the Rendezvous, Track, First Pass, and Update flags and resets the LOSCM, R04, Manual Acquire, No Angle Monitor, and Search flags. The Rendezvous flag is set to denote that program P-20 or P-22 is being used. When this flag is reset, programs P-20 and P-22 are terminated. The purpose of most of the other flags is given in the following sections. In addition to initializing the above flags, the state vector update option is automatically set to the LM in program P-20.

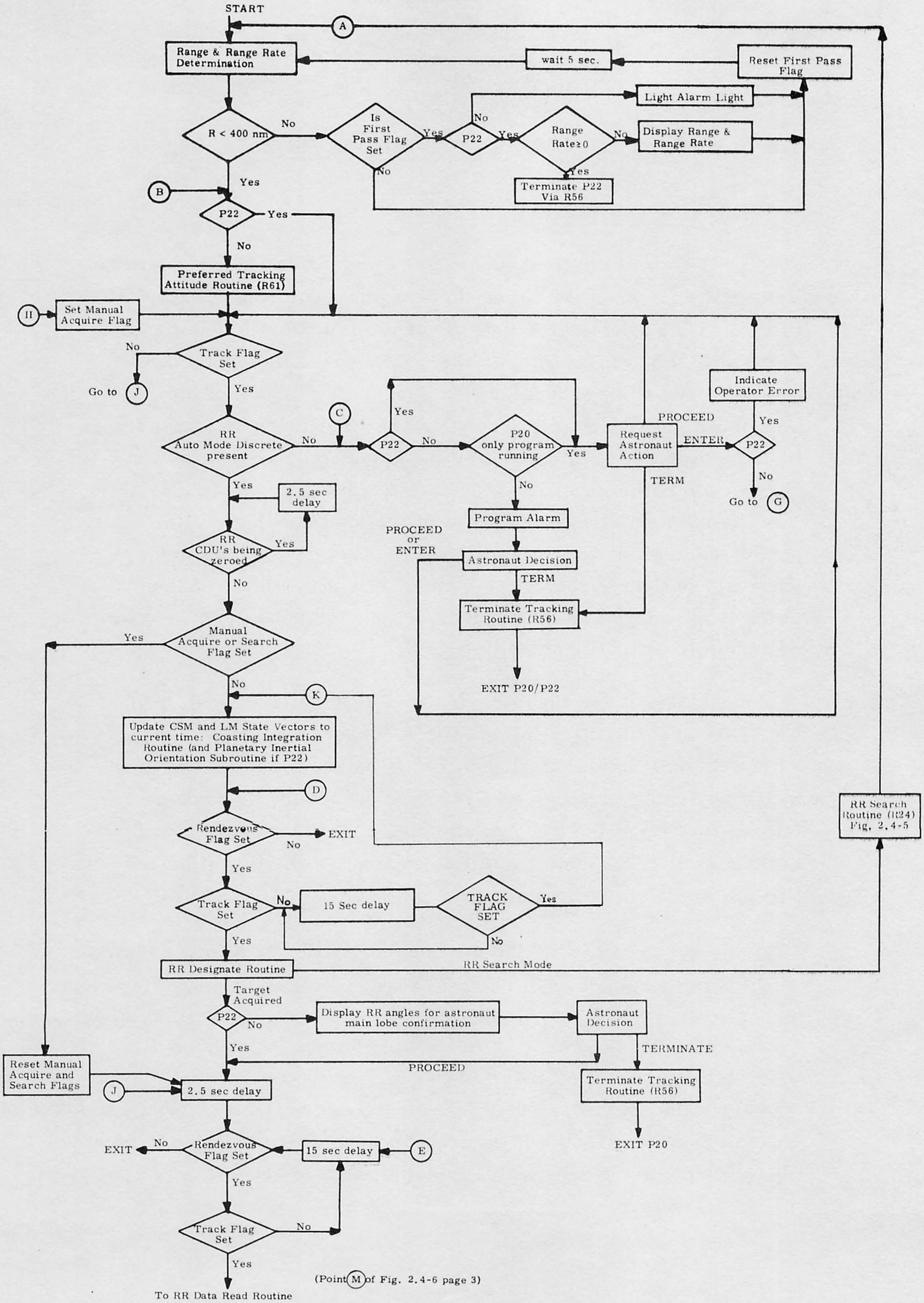


Figure 2.4-1. Target Acquisition Routine (Page 1 of 2)

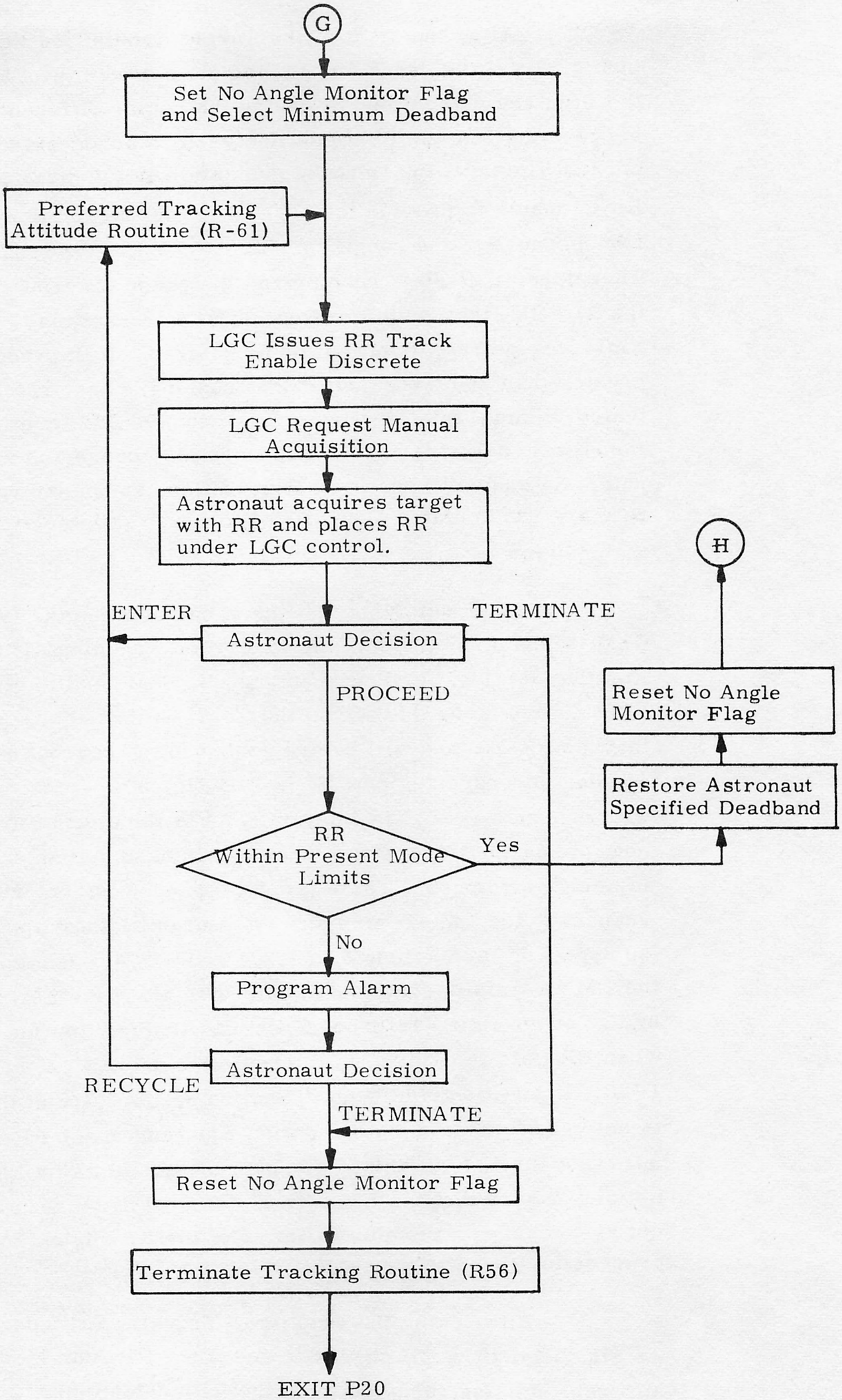


Figure 2.4-1 Target Acquisition Routine
(page 2 of 2)

After the start of the Target Acquisition Routine in Fig. 2.4-1 the range and range rate between the LM and the CSM are determined by taking the vector difference between the LM and CSM position vectors propagated to current time. If the range is greater than 400 n.mi., the RR is unable to provide the correct range information to the LGC because of the ranging technique used in the radar. Therefore, if P-20 is running and the range exceeds 400 n.mi., the ALARM light is illuminated. The astronaut may call the range, range rate display if he so desires. If the range is greater than 400 n.mi. and P22 is running, range rate is tested. Range rate positive or zero causes P22 termination. Range rate negative causes P22 to display range and range rate. So long as there is no termination, range and range rate are recalculated every five seconds, until range is less than 400 n.mi.

After verifying that the range between the LM and CSM is less than 400 nm, the Preferred Tracking Attitude Routine (R-61 of Sections 4 and 5.2.4.4) is used to align the LM +Z-axis along the LOS to the CSM. This is done to insure that the RR antenna will be designated in the correct antenna angular coverage region (Mode 1 of Fig. 2.4-3) for operation with program P-20. In addition, this attitude permits the LM optical beacon to be seen by the CSM in case optical tracking is being performed. The LM optical beacon is centered with respect to the LM +Z-axis and has a beamwidth of approximately 60 degrees. In Sections 5.2.4.2.1 and 5.2.4.4 it is seen that the LM +Z-axis is continuously directed along the LOS to the CSM by use of the Fine Preferred Tracking Attitude Routine (R-65) when RR Data is being used to update the navigation equations in P-20. In addition it is seen in Section 5.2.4.2.1 that the RR data is not used to update the navigation equations if the RR is more than 30^{o} from the LM +Z-axis. The above constraints on LM attitude not only insure that the LM optical beacon will be seen by the CSM but also insure a reliable estimate of the RR angle biases when processing RR data.

After using the Preferred Tracking Attitude Routine in Fig. 2.4-1 the LGC checks to see if the RR Auto Mode discrete is being received from the RR. This discrete signifies that the RR is on and has been placed under LGC control by a

mode control switch associated with the RR. In addition, reception of this discrete at this time automatically indicates that the astronaut wishes to use the RR LGC Mode of target acquisition. If the discrete is not present and P-20 is the only program in operation, a request is made to the astronaut to either select the RR Manual Mode by keying in "ENTER" or the RR LGC Mode by placing the RR mode control switch in the LGC position and keying in "PROCEED." If program P-20 is not the only program running, a program alarm is issued to the astronaut requiring the action indicated in Fig. 2.4-1. This latter step is taken to avoid conflict in DSKY displays between program P-20 and other programs.

5.2.4.1.1 Target Acquisition with the RR LGC Mode

If the check on the RR Auto Mode discrete in Fig. 2.4-1 indicates that the RR LGC Mode was selected, the routine next insures that the RR CDU's are not being zeroed. These CDU's are zeroed by the RR Monitor Routine (R-25) of Section 5.2.4.3 whenever the RR Auto Mode discrete is received from the RR. Afterwards, a check is made to see if either the Manual Acquire or Search flag is set, signifying that the RR Manual or RR Search Mode is being used. Since the present mode is assumed to be the RR LGC Mode, these flags should not be set. Just before entering the RR Designate Routine, a precision update is made on the CSM and LM permanent state, and the Rendezvous and Track flags are checked. The precision update is performed at this time in order to shorten the operation time of the Kepler Subroutine, which is used periodically in the RR Designate Routine to compute the LOS to the CSM. The operation time of the Kepler Subroutine is a function of the age of the permanent state vector. The purpose of the check on the Track flag just prior to using the RR Designate Routine is to insure that the RR is not designated by the LGC until this flag is set. The Track flag is reset by various LGC programs whenever there is no desire to designate or read data from the RR. Although the Track flag is set at the start of program P-20, it is possible for it to be temporarily reset during operation of program P-20.

To designate the RR along the LOS to the CSM, use is made of the RR Designate Routine of Fig. 2.4-2. Initially the RR Track Enable discrete is removed from the RR in order to ensure RR response to designate commands. Subsequently, the routine points the RR antenna along the reference direction of Mode 1 for P-20 (see Fig. 2.4-3). For P-22, a check is made to see if the antenna is in Mode 2, as defined in Section 5.2.5.3, and, if not, it is pointed along the reference direction for Mode 2. The LOS range vector ($\underline{r}_{LOS}$) and the velocity ($\underline{v}_{LC}$) of the CSM with respect to the LM in stable member coordinates are computed by using the CSM and LM position and velocity vectors ($\underline{r}_C$, $\underline{v}_C$, $\underline{r}_L$, $\underline{v}_L$). During P-20 the vectors $\underline{r}_C$, $\underline{v}_C$, $\underline{r}_L$, and $\underline{v}_L$ are obtained by using the Kepler Subroutine of section 5.5.5. During P-22 only $\underline{r}_C$ and $\underline{v}_C$ are obtained with the Kepler Subroutine -- $\underline{r}_L$ and $\underline{v}_L$ are obtained in the precision update prior to entering RR Designate Routine. The vectors $\underline{r}_L$ and $\underline{v}_L$ are not updated during operation of the routine for P-22, because the update is time consuming and, due to the slow rate of lunar motion, would result in very little improvement in the LOS accuracy. The specified time for use in the Kepler Subroutine is the present time plus an additional amount ϵ_1. The quantity ϵ_1 is equal to 0.5 seconds and is a rough estimate of the LGC time required by the Kepler Subroutine to compute $\underline{r}_C$ and $\underline{v}_C$ during the operation of P-22. The main reason for using ϵ_1 is to ensure that the vector $\underline{r}_{LOS}$ issued to the gyro command loop (see page 2 of Fig. 2.4-2) is not too stale in time during the relatively large LOS angular rates experienced in P-22. This time correction is used in P-20 also, to offset some of the time staleness. The correction is not as good for P-20, but the problem is not as acute because of the relatively low LOS angular rates experienced during free-fall.

After computing $\underline{r}_{LOS}$ and $\underline{v}_{LC}$, a check is made (Fig. 2.4-2) to see if the LOS is within the angular tracking limits of either RR antenna mode shown in Fig. 2.4-3. This is accomplished by first computing the equivalent RR shaft and trunnion angles for $\underline{r}_{LOS}$ with the method shown in Section 5.6.15.2 where

$$\underline{u}_D = [SMNB]\, UNIT(\underline{r}_{LOS})$$

and then comparing these angles with the electrical tracking mode limits. It should be noted that the tracking mode limits of Fig. 2.4-3 are not the true LOS limits of the RR antenna modes but are the LOS limits within which the RR should track satisfactorily. The polarity and magnitude of the shaft and trunnion LOS values given in Fig. 2.4-3 agree with the corresponding shaft and trunnion CDU indicated values except for shaft angles in Mode 2 where

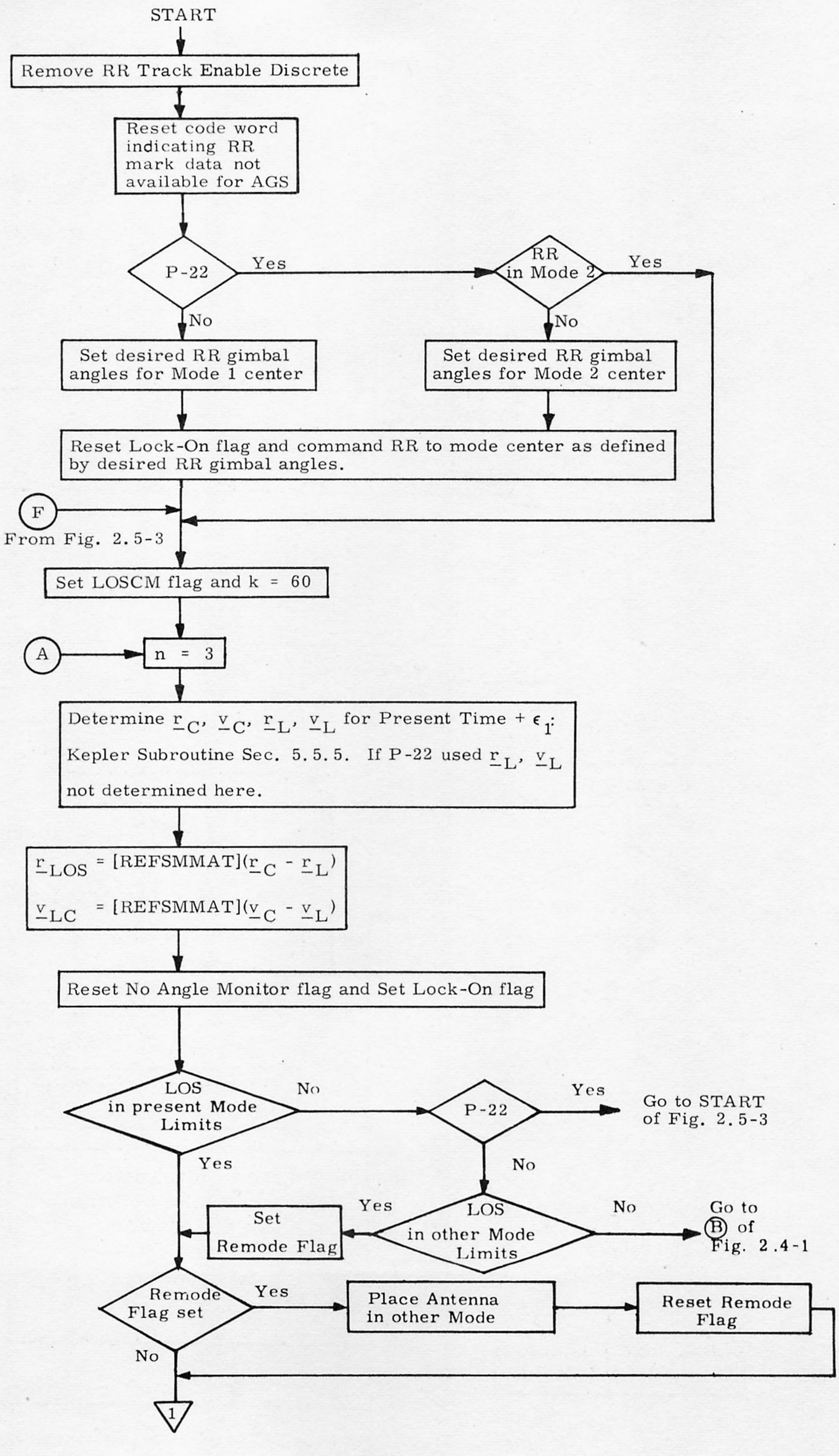


Figure 2.4-2 RR Designate Routine (Page 1 of 2)

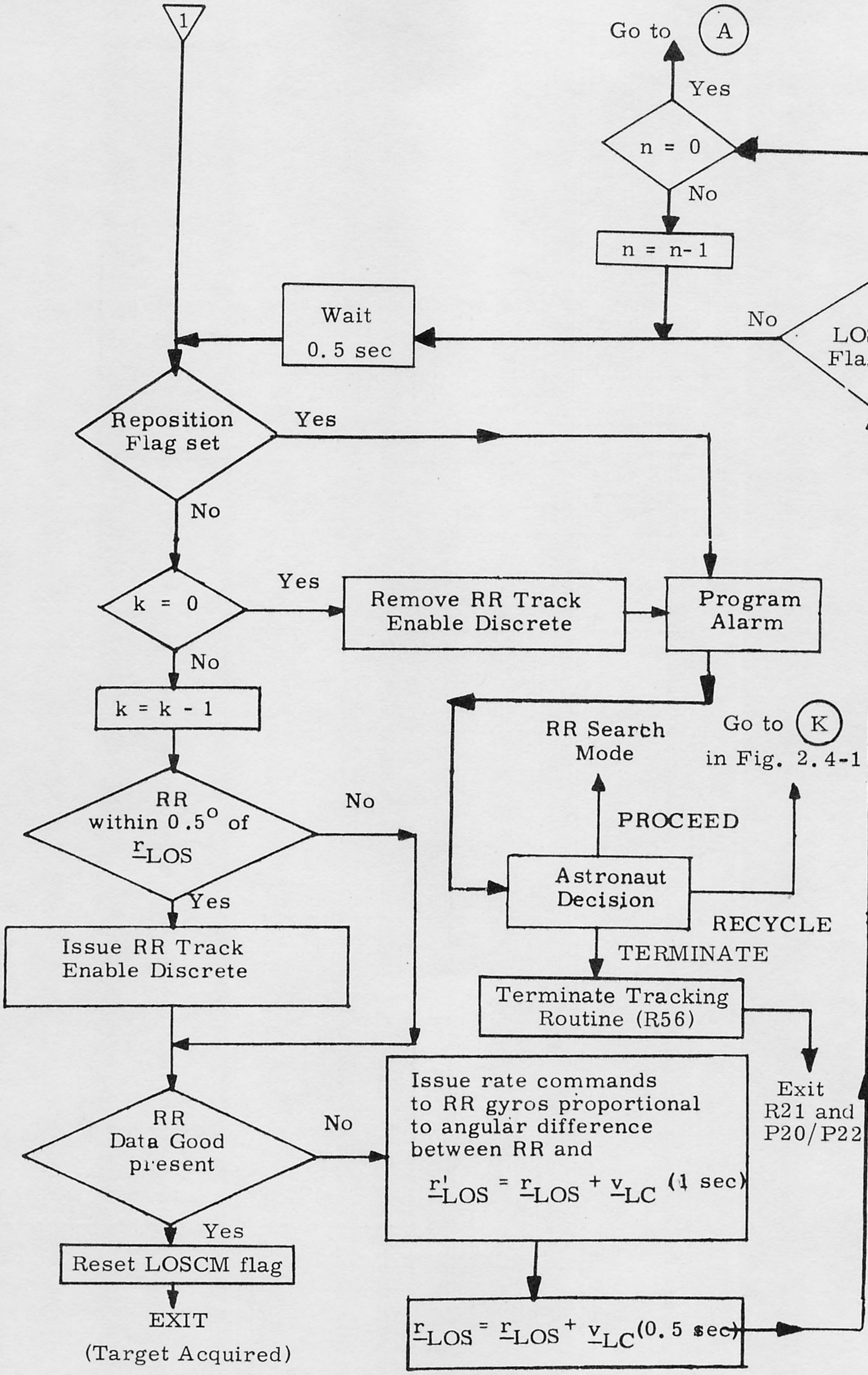


Figure 2.4-2 RR Designate Routine (Page 2 of 2)

$$S_{CDU} = S_{LOS} + 180°$$

The RR Monitor Routine (R-25 of Section 5.2.4.3) continually updates the computer's knowledge of which mode the RR antenna is actually in, even though the antenna may not be within the tracking limits of that mode. Since the Preferred Tracking Attitude Routine was previously used in program P-20 to align the LM +Z-axis along the LOS, it is seen in Fig. 2.4-2 that steps are taken to insure that the RR antenna is in Mode 1 and the LOS is within the tracking limits of Mode 1 before designating the RR. Whenever a change in antenna mode is performed (i.e. a remode) the antenna is left pointing along the reference direction for the desired mode as shown in Fig. 2.4-3.

If the Reposition flag should be set by the RR Monitor Routine of Section 5.2.4.3 during the operation of the gyro command loop in the RR Designate Routine, a program alarm is issued as shown in Fig. 2.4-2. This flag when set denotes that the RR is being repositioned to the reference direction for the present RR antenna mode.

The RR is designated towards the CSM by issuing rate commands to the RR antenna gyros approximately every 0.5 seconds which are proportional to the angular difference between the indicated direction of the antenna and

$$\underline{r}'_{LOS} = \underline{r}_{LOS} + \underline{v}_{LC} \text{ (1 sec.)}$$

where $\underline{r}'_{LOS}$ is essentially $\underline{r}_{LOS}$ advanced one second into the future and is obtained by adding to the present range vector ($\underline{r}_{LOS}$) the distance covered in a one second interval by $\underline{v}_{LC}$ assuming a constant velocity. This correction to $\underline{r}_{LOS}$ for RR designation essentially compensates for the lag error associated with the type of control system existing between the RR and the LGC. The method used to compute the gyro rate commands is given in Section 5.6.15.3.

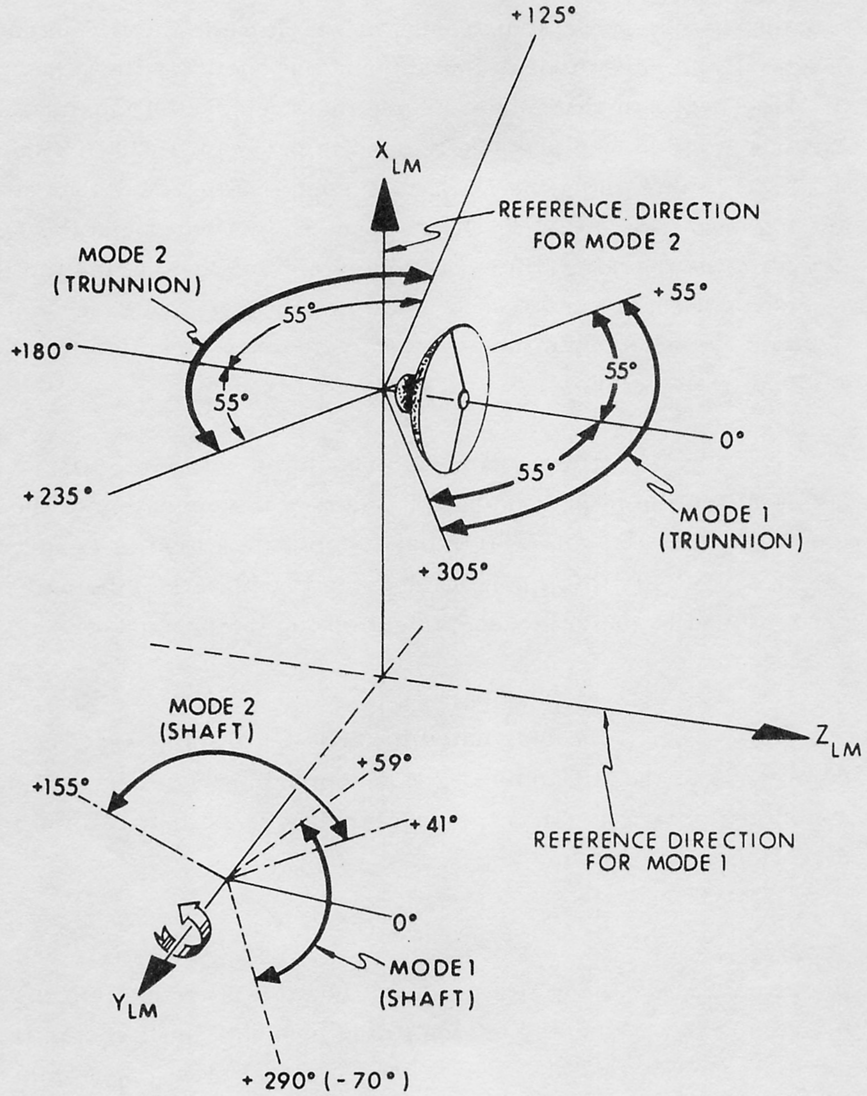


Figure 2.4-3 RR Antenna Shaft and Trunnion LOS Tracking Regions

In Fig. 2.4-2 it is seen that an approximate update for LOS motion is made to $\underline{r}_{LOS}$ [i.e. $\underline{r}_{LOS} = \underline{r}_{LOS} + \underline{v}_{LC}$ (0.5 sec)] about every 0.5 seconds until the counter n has been decremented from 3 to 0. When n = 0 the routine returns to the top of Fig. 2.4-2 to compute new values of $\underline{r}_{LOS}$ and $\underline{v}_{LC}$.

During the designation of the RR a check is made to see if the RR is within 0.5 degrees of the present LOS range vector $\underline{r}_{LOS}$. This is accomplished by using the method in Section 5.6.15.1 to obtain a unit vector $\underline{u}_{RR}$ defining the direction of the RR in navigation base coordinates, which is then transformed to stable member coordinates as follows:

$$\underline{u}_{RR} = [NBSM]\ \underline{u}_{RR}$$

and compared with the present LOS range vector $\underline{r}_{LOS}$. When the angle between $\underline{u}_{RR}$ and $\underline{r}_{LOS}$ drops below 0.5 degrees, the RR Track Enable discrete is issued to the RR, enabling its angle tracking servos to track the target if its range rate tracking network has already acquired the target. Issuance of the RR Track Enable discrete also initiates the RR range tracker search.

Approximately every 0.5 seconds it is seen in Fig. 2.4-2 that a check is also made to see if the RR Data Good discrete is present. This discrete is sent to the LGC by the RR when lock-on has been achieved in range and range rate and the RR Track Enable discrete has been received from the LGC. In the event of failure to receive this discrete after issuing rate commands to the RR gyros 60 times, the RR Track Enable discrete is removed and a program alarm is issued to the astronaut whereupon he either repeats the designate process or goes to the RR Search Mode.

If the RR Data Good discrete is received during the designate process, the routine is terminated and it is seen in Fig. 2.4-1 that for P-20, the angles indicating the direction of the RR antenna are displayed to the astronaut so that he may confirm lock-on by the main radiation lobe before entering the RR Data Read Routine. This confirmation may be made through use of the Crew Optical Alignment Sight (COAS) or the RR signal-strength meter. The angles in the display are the same as those defined in Sections 5.6.14 and 5.6.21. If the astronaut should exercise manual control of RR to insure that it is tracking on the main lobe, he should not key in a PROCEED until after he has placed the RR mode control switch in the LGC position and the radar panel NO TRACK light is extinguished. This procedure is necessary for the reasons given in Section 5.2.4.1.2. After confirming main lobe lock-on, a 2.5 second delay is introduced in order to permit any transients in the RR angle tracking servos to settle out before RR data is taken. In addition, the status of the Rendezvous and Track flags is checked before entering the RR Data Read Routine of Fig. 2.4-6.

5.2.4.1.2 Target Acquisition With the RR Manual Mode

In Fig. 2.4-1 it is seen that the RR Manual Mode of target acquisition can be obtained by keying in "ENTER" after the LGC discovers that the RR Auto Mode discrete is not present. Absence of this discrete at the beginning of program P-20 can be insured by not placing the mode control switch of the RR in the LGC position. The logic associated with the RR Manual Mode is shown on page 2 of Fig. 2.4-1. Initially, the No Angle Monitor flag is set, the minimum deadband of the RCS DAP is selected, and the RR Track Enable discrete is issued to the RR. The No Angle Monitor flag is set during the RR Manual Mode so as to disable the angle monitor function of the RR Monitor Routine (see Section 5.2.4.3). Selection of the minimum deadband permits the astronaut to manually designate the RR more accurately. If the Preferred Tracking Attitude Routine (R-61) is called by keying RECYCLE and an R-60 maneuver is called, the astronaut-specified deadband is restored at the conclusion of the maneuver. The RR Track Enable discrete, although it has no effect on manual control of the RR, is issued at this time so that there is no loss of RR angle tracking when the mode control switch of the RR is placed in the LGC position after manual target acquisition.

Afterwards, the LGC requests the astronaut to perform the manual acquisition. If manual acquisition is achieved, the astronaut places the RR under LGC control, waits until the radar panel NO TRACK light is extinguished, and keys in "PROCEED". This procedure is necessary in order to insure that the RR range tracking network has locked onto the target before entering the RR Data Read Routine. When the RR mode control switch is placed in the LGC position RR range tracking is interrupted by switching and a new range search is initiated. When the RR acquires the target in both range and range rate, the NO TRACK light is extinguished.

After the "PROCEED" the LGC checks to see if the RR is within the limits of the present coverage mode. If the RR is not within the coverage limits, a program alarm is issued and the astronaut either terminates the program or repeats the manual acquisition process as shown in Fig. 2.4-1. Note that the Preferred Tracking Attitude Routine is used to re-align the LM +Z-axis with the target LOS whenever he elects to repeat the manual acquisition process.

Once target acquisition has been achieved and the RR is found to be within the present antenna mode limits, the astronaut specified deadband is restored, the No Angle Monitor flag is reset, and the Manual Acquire flag is set to denote manual acquisition. The remaining steps are indicated in Fig. 2.4-1.

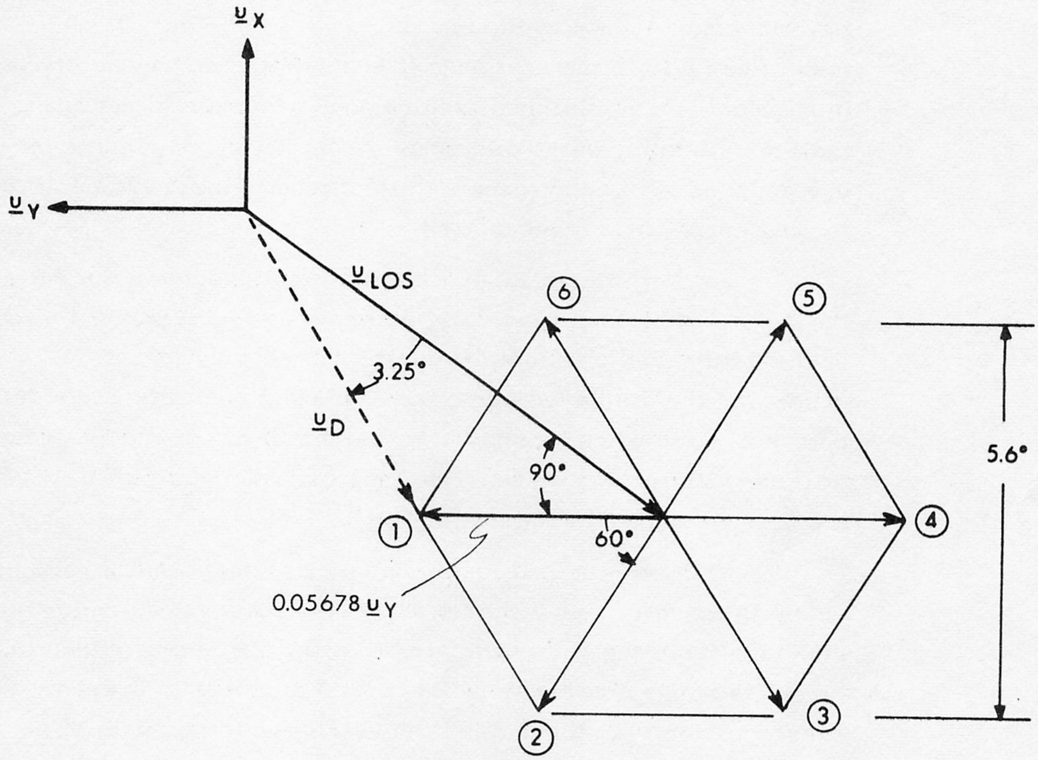
5.2.4.1.3 Target Acquisition With the RR Search Mode

In the RR Designate Routine (Fig. 2.4-2) it is seen that the astronaut may select the RR Search Mode if the RR Designate Routine fails to acquire the target. The RR Search Mode is obtained by using the RR Search Routine (R-24 of Section 4) to designate the RR antenna in a hexagonal search pattern about the estimated LOS. The search pattern is a box with six sides where the side to side dimensions are 5.6^o as shown in Fig. 2.4-4. At the beginning of this mode the RR is designated for six seconds along the estimated LOS to the target defined by the unit vector $\underline{u}_{LOS}$ in Fig. 2.4-4. Afterwards, the LGC sequentially designates the RR to each corner of the hexagon for a period of six seconds. Having completed the designate to each corner, the LGC repeats the above process starting with a designate along $\underline{u}_{LOS}$ for six seconds. The time required to generate this search pattern is approximately 42 seconds.

The logic flow associated with the RR Search Routine is shown in Fig. 2.4-5. Initially, the Search flag is set to denote that this mode is being used and a display of certain RR search parameters is instigated to which the astronaut must respond later. The RR Track Enable discrete is then issued to the RR so that the RR may acquire the target during the search pattern generation. This discrete is also re-issued at each corner of the search pattern in case it has been removed by some source such as the RR Monitor Routine.

Approximately every six seconds the position and velocity vectors of the CSM and LM are used to compute the line-of-sight unit vector ($\underline{u}_{LOS}$) in basic reference coordinates and the relative velocity ($\underline{v}_{LC}$) in stable member coordinates. Note that $\underline{u}_{LOS}$ and $\underline{v}_{LC}$ are based on the CSM and LM position and velocity vectors computed for the present time plus ϵ_2 where ϵ_2 is equal to 1.5 seconds and is a rough estimate of the LGC time required to compute the CSM and LM state vectors during the operation of P-22. The reason for using ϵ_2 is to ensure

that the vector $\underline{u}_{LOS}$ is not too stale in time during the relatively large LOS angular rates experienced in P-22. After computing $\underline{u}_{LOS}$ and $\underline{v}_{LC}$, the routine computes the desired RR pointing direction ($\underline{u}_D$) which may be



Note: This is the search pattern as viewed from the CSM.

Fig. 2.4-4 RR Search Pattern

along $\underline{u}_{LOS}$ or to one corner of the search pattern (see Figure 2.4-4). Afterwards, $\underline{u}_D$ is transformed to stable member coordinates and a check is made on the direction of $\underline{u}_D$ with respect to the angular coverage modes of the RR antenna, just as was done with $\underline{r}_{LOS}$ in the RR Designate Routine in Section 5.2.4.1.1. If $\underline{u}_D$ is not within the angular coverage limits of either mode during program P-20, the program alarm light is turned on, an alarm code is stored, and the search pattern is stopped. If the astronaut wishes to continue the search in program P-20, he must re-establish the preferred tracking attitude with the Preferred Tracking Attitude Routine (R-61) in the manner shown at the end of Fig. 2.4-5. Once $\underline{u}_D$ is found to be within the coverage limits and the correct antenna mode has been established, the routine proceeds to designate the RR by issuing rate commands to the RR gyros about every 0.5 seconds with approximate corrections being made each time for lag error and target motion.

Note in Fig. 2.4-5 that the angle between the RR and the LM +Z-axis is periodically determined and displayed as one of the RR search parameters during the search operation. By observing the displayed angle the astronaut can determine during program P-20 when he should re-establish the preferred tracking attitude with the Preferred Tracking Attitude Routine (R-61) in the manner shown at the end of Fig. 2.4-5.

Near the end of Fig. 2.4-5 it is seen that a periodic check is made to see if the RR Data Good discrete is being received from the RR, signifying that the RR has acquired the target in range and range rate. If this discrete is present, the search pattern is stopped and the astronaut is notified. The astronaut then checks to see if acquisition was obtained with the main radiation lobe of the RR. By manually positioning the RR and observing the RR signal strength meter, he should be able to distinguish the main lobe from any side lobes. Having achieved and verified lock-on with the main radiation lobe, the astronaut places the RR mode control switch in the LGC position, waits until the radar panel NO TRACK light is extinguished, and keys in a "PROCEED". The remaining steps are indicated in Fig. 2.4-1.

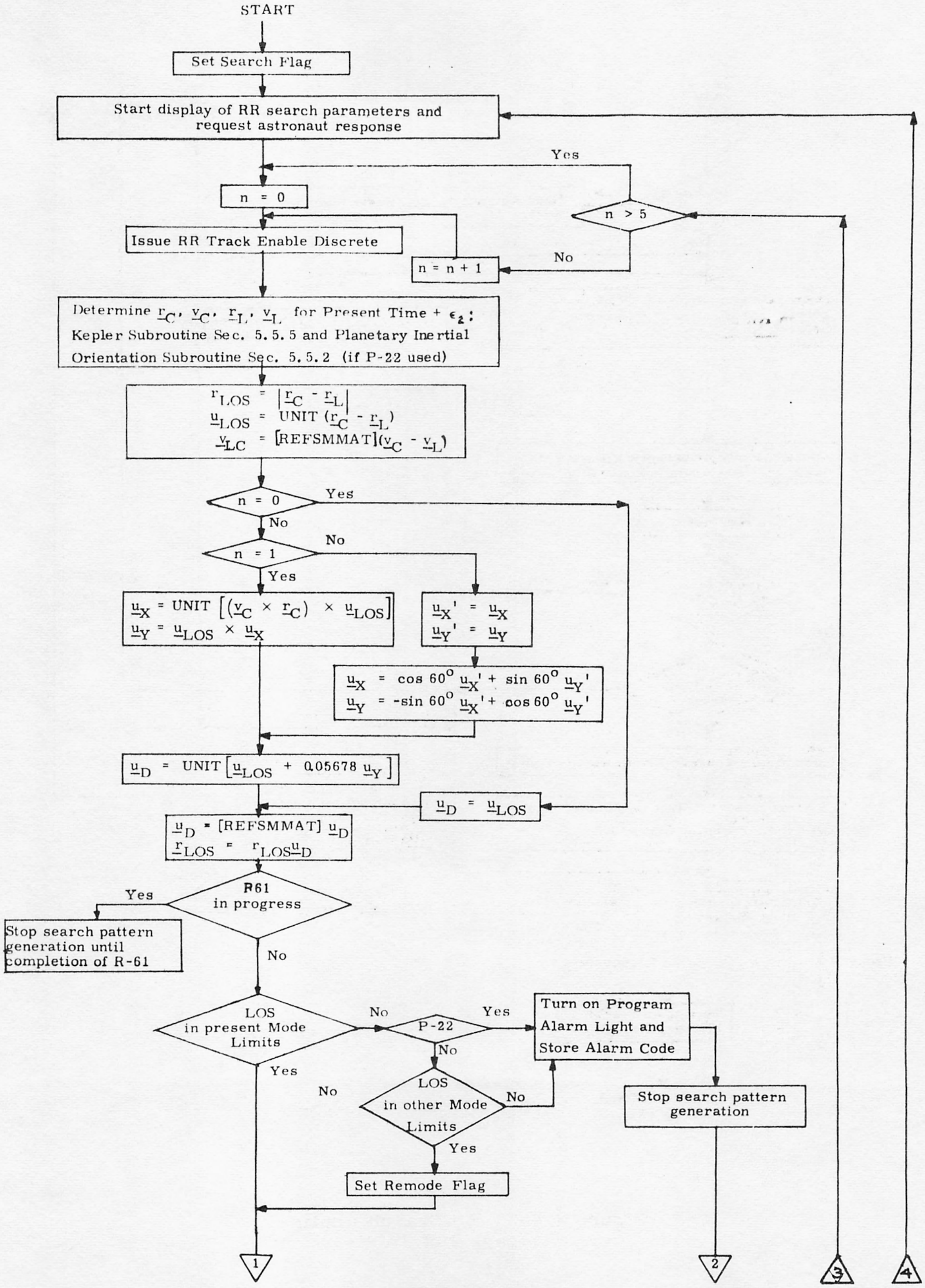


Figure 2.4-5 RR Search Routine (page 1 of 2)

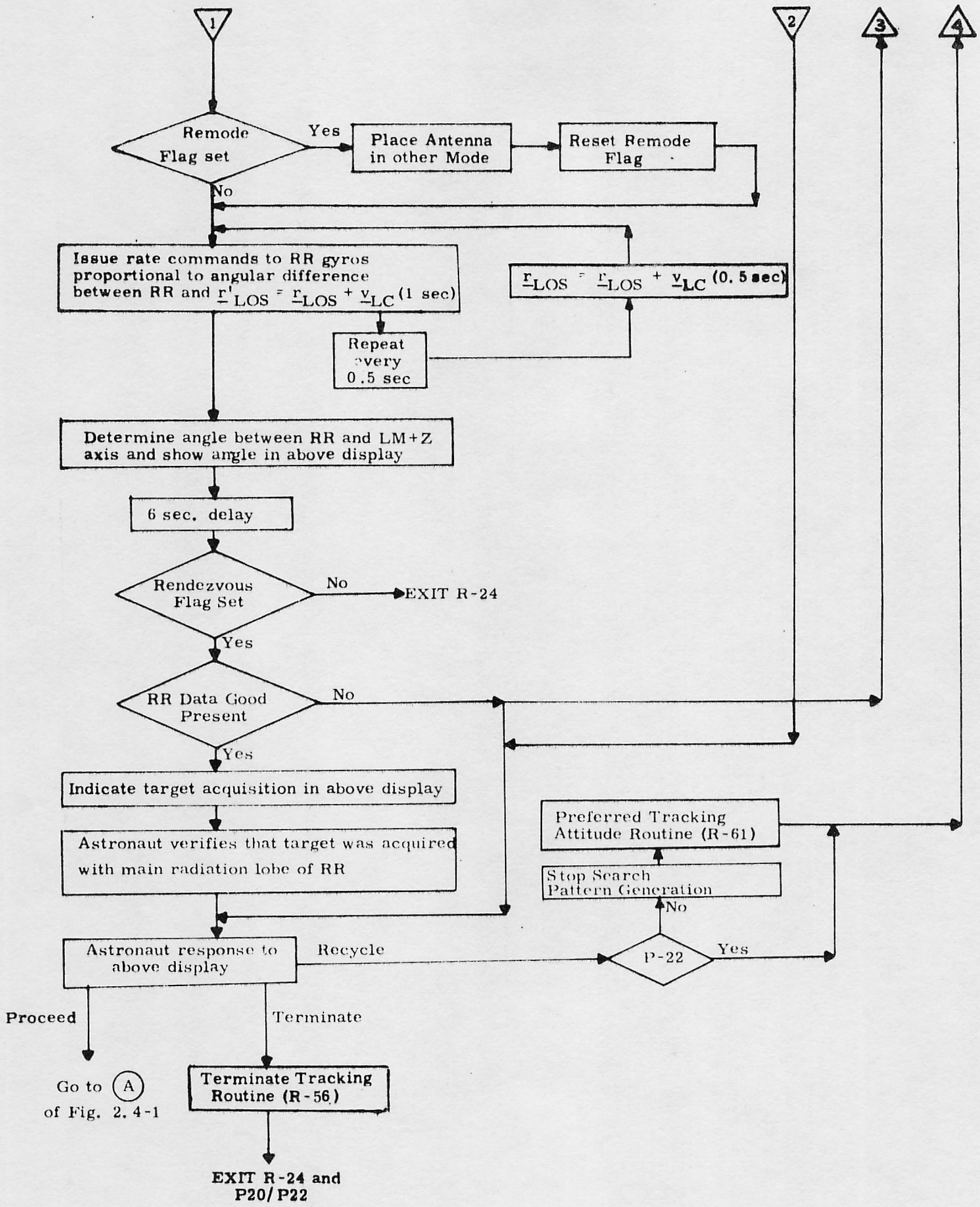


Figure 2.4-5 RR Search Routine
(page 2 of 2)

5.2.4.2 Rendezvous Navigation Routine

5.2.4.2.1 RR Data Read Routine

During operation of the Rendezvous Navigation Routine use is made of the RR Data Read Routine (R-22 of Section 4) to obtain measurement data from the RR. The logic associated with the RR Data Read Routine is given in Fig. 2.4-6. Like the Target Acquisition Routine, this routine is used by both the Rendezvous and Lunar Surface Navigation Programs, with different paths being taken at various points in the routine depending on which program (P-20 or P-22) is in operation. The RR Data Read Routine periodically obtains a complete set of data from the RR (range, range rate, shaft angle, and trunnion angle) for purposes of navigation, although only range and range rate data is used for updating during lunar surface navigation. When routine R-22 is used in program P-20, the Fine Preferred Tracking Attitude Routine (R-65 of Sections 4 and 5.2.4.4) is called on a repetitive basis to obtain continuous or fine LM +Z-axis tracking of the CSM. Most of the details on continuous Z-axis tracking in program P-20 are given in Section 5.2.4.4. During the operation of program P-20 the maximum frequency of update of the navigation equations with a complete set of RR data is about 64 seconds. This is based upon a rough estimate given in Section 5.2.4.4.

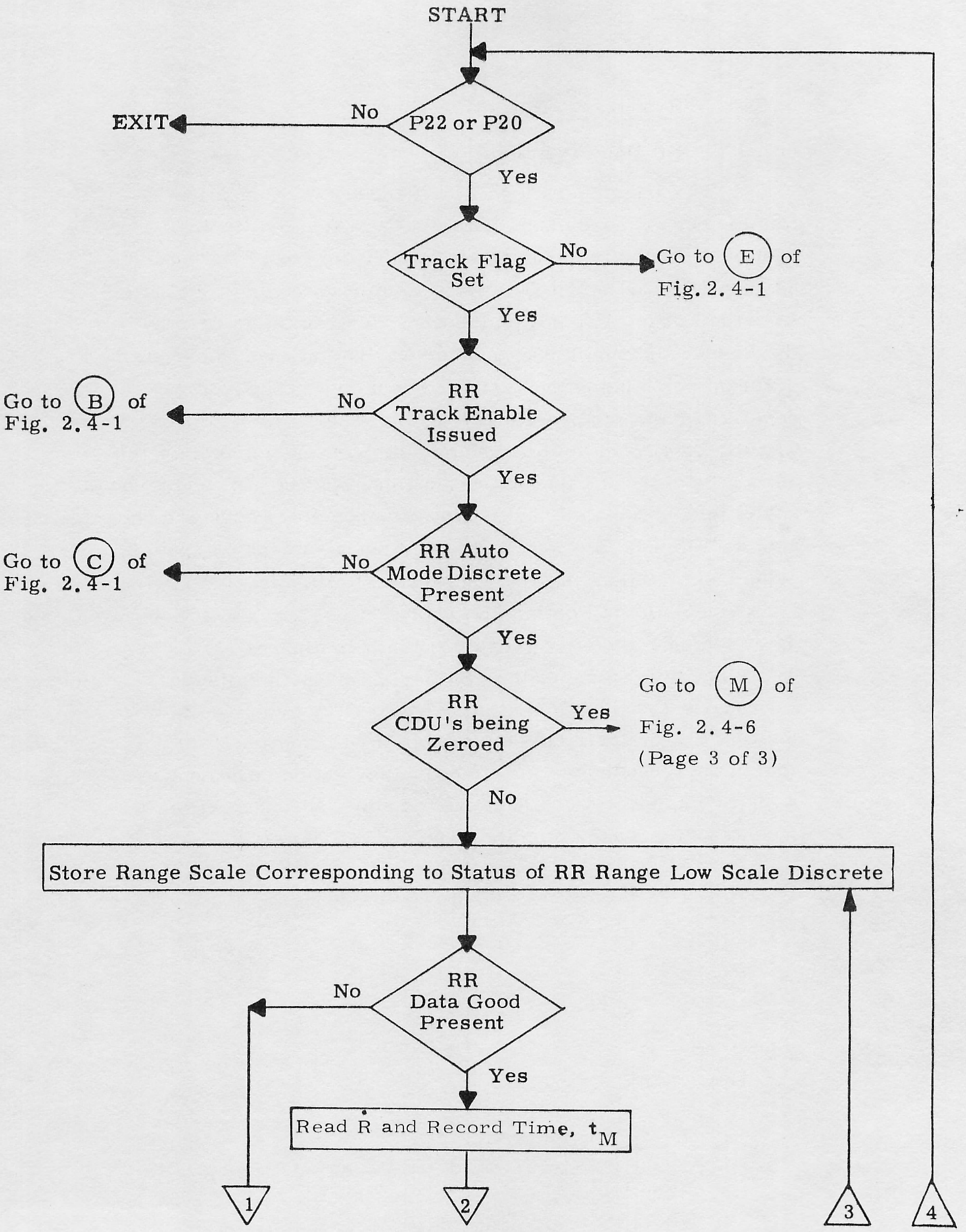


Figure 2.4-6 RR Data Read Routine (Page 1 of 3)

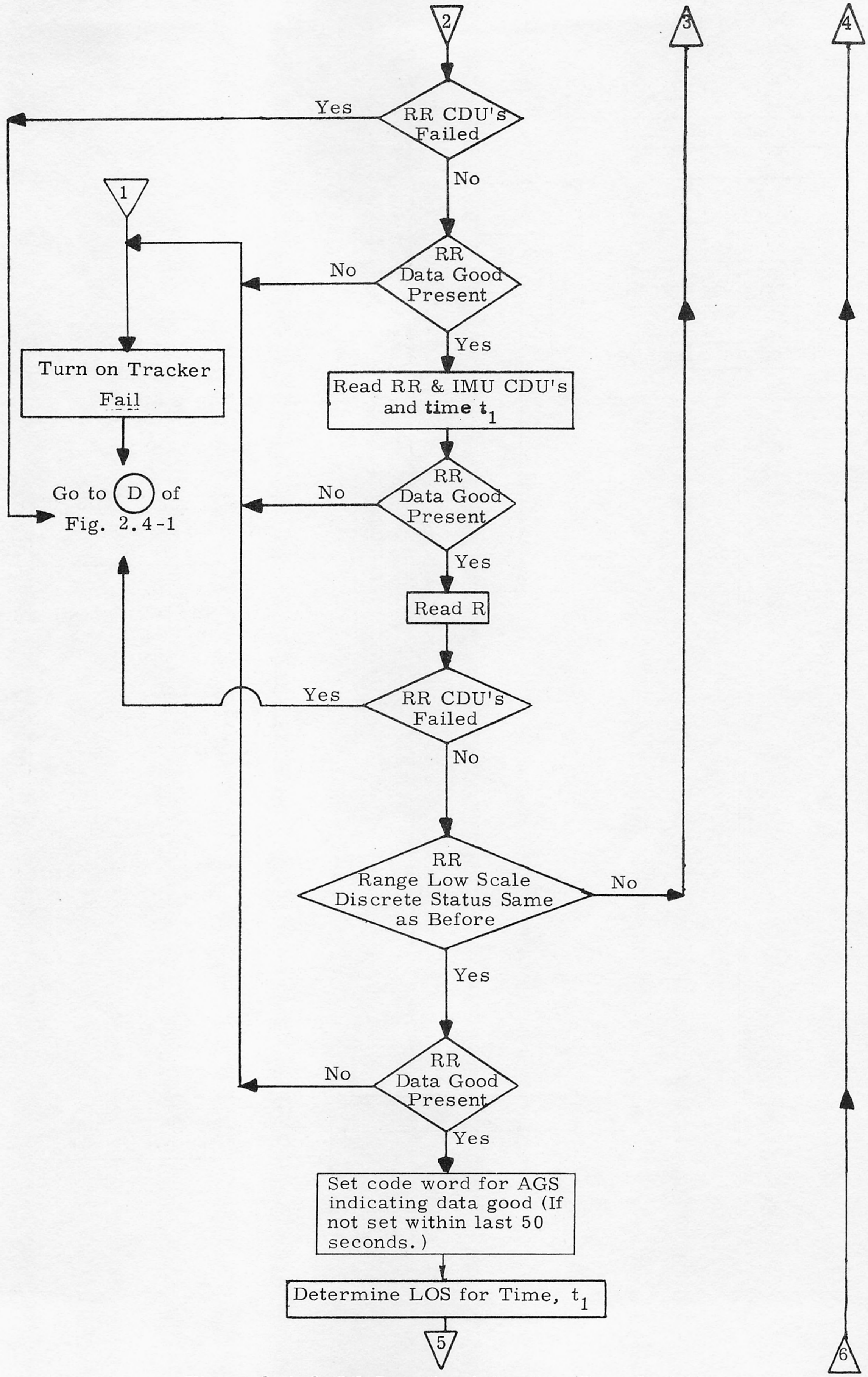


Figure 2.4-6 RR Data Read Routine (Page 2 of 3)

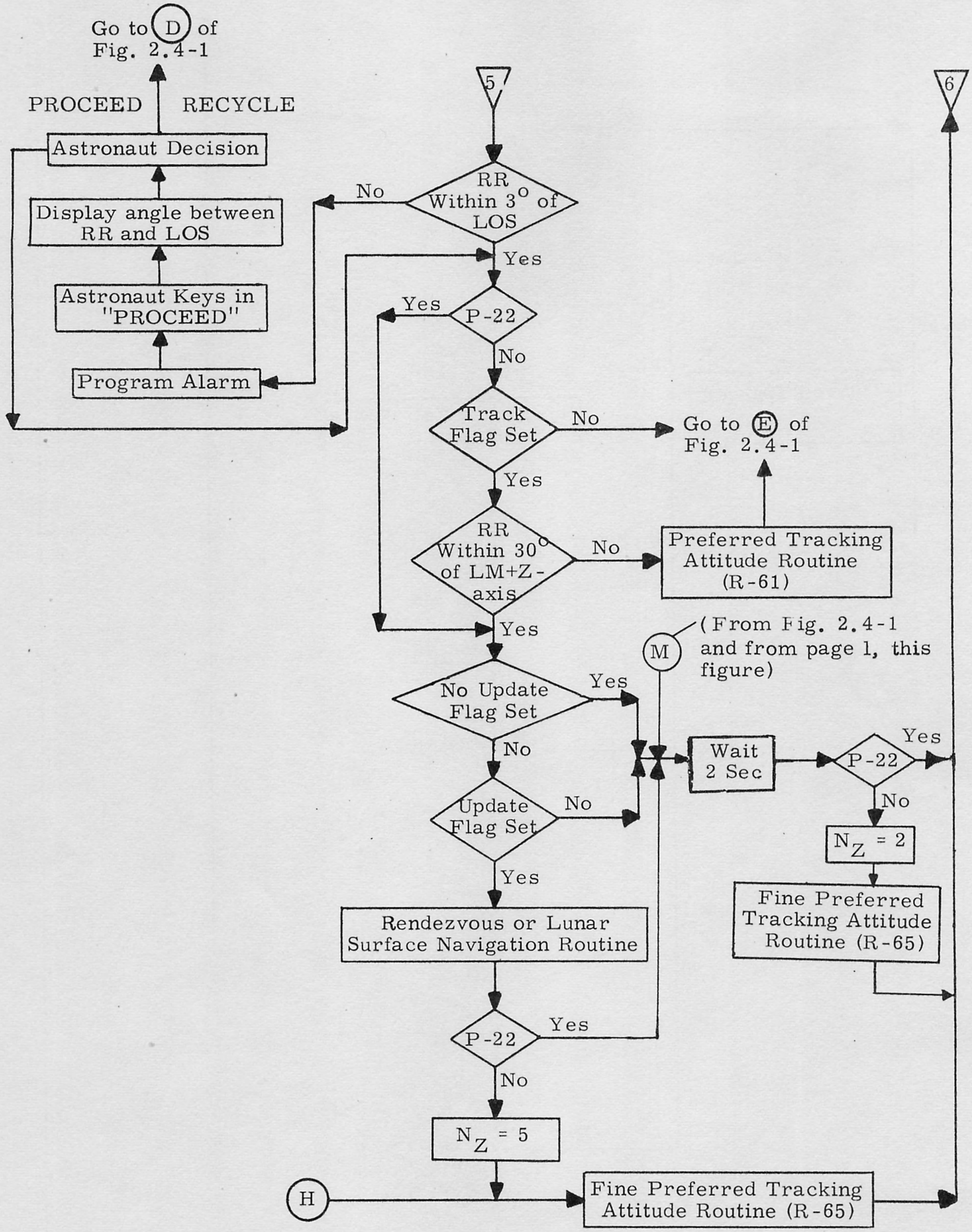


Figure 2.4-6 RR Data Read Routine
(Page 3 of 3)

In this section an explanation will be given only for those logic steps in Fig. 2.4-6 associated with the use of the RR Data Read Routine during operation of the Rendezvous Navigation Program (P-20).

In Fig. 2.4-6, the first check made by the routine, after establishing that P-20 or P-22 is being used, is to see if the Track flag is present. This flag is removed during the preparation and execution of a LM ΔV maneuver when there is no desire to have the routine request RR data or call any other routine such as the RR Designate and Fine Preferred Tracking Attitude Routines.

If the Track flag is present, the routine next checks for issuance of the RR Track Enable discrete and reception of the RR Auto Mode discrete. The former discrete is removed and not re-issued by the RR Monitor Routine whenever the RR antenna angles exceed the limits in Fig. 2.4-3. Its absence therefore indicates a need to go back to input B of Fig. 2.4-1 to re-establish the preferred tracking attitude and re-designate the radar. Afterwards, a check is made to insure that RR data is not taken if the RR CDU's are being zeroed. If the RR CDU's are being zeroed and program P-20 is in operation, the Fine Preferred Tracking Attitude Routine (R-65) of Section 5.2.4.4 is called with a specified value of 2 for the quantity N_Z, which indicates how many times routine R-65 is to repeat itself before returning to routine R-22.

The sequence used by the routine for reading RR data is shown in Fig. 2.4-6 where frequent checks are made on the RR Data Good discrete to insure that no RR tracking interruptions have occurred during the read-out. If this discrete is missing, the Tracker Fail light is turned on before going to point D of Fig. 2.4-1 to re-designate the RR. Note in Fig. 2.4-1 that checks are also made to see if the RR CDU's have failed during the read-out and, if so, a return will also be made to point D of Fig. 2.4-1.

Prior to and after reading the data, the routine checks the status of the Range Low Scale discrete to insure that the proper scaling is applied to the range data. For ranges below 9.38 × 32,767 feet, the RR issues the Range Low Scale discrete indicating to the LGC that the low scale factor should be used. If the status of this discrete should change during the data read process, it is seen in Fig. 2.4-6 that a new data request is immediately made by the routine.

It should be noted that a single time tag t_M is used for a complete set of RR data (range, range rate, shaft angle, and trunnion angle) in the navigation computations. This time (t_M) is the time recorded at the middle of the 80 millisecond counting interval for the range rate measurement.

Whenever RR data is successfully read, a code word is set to indicate to the AGS that RR data is available on the downlist, provided that it has been at least 50 seconds since the last setting.

After the RR data is read, the LOS is determined for time t_1 and compared with that indicated by the RR CDU's. This is done to insure against RR side-lobe lock-on. If the difference is more than 3 degrees, an alarm is issued to the astronaut who must then decide to go to the RR Designate Routine or request the LGC to proceed with the data.

If the Track flag is still present after the side-lobe check, the routine checks to see if the angle between the RR antenna and the LM +Z-axis is within 30 degrees. As mentioned previously, this is required in order for the LGC to perform a satisfactory determination of RR angle biases, and to also enable the LM Optical Beacon to be seen by the CSM. If the antenna is not within 30 degrees of the LM +Z-axis, the Preferred Tracking Attitude Routine (R-61) is called before returning to E of Fig. 2.4-1.

Afterwards, the LGC checks the Update and No Update flags. The Update flag is reset by various programs and routines whenever it is desired to temporarily stop the update of the navigation equations with RR data. The No Update flag is set by the astronaut whenever he wishes to permanently stop the update of the navigation equations with RR data. This flag is reset whenever the astronaut indicates that he wishes to update either the LM or CSM state vector. It should be noted that it is only necessary for the Track flag to be present in order to monitor RR tracking and read RR data even though the data may not be used for navigation updates.

The range obtained from the RR by the RR Data Read Routine is that measured by the RR between the LM and the CSM. This data is sent to the LGC from the RR as a binary data word R_{RR}. In the LGC the range r_{RR} in feet is obtained as follows:

$$r_{RR} = \begin{cases} k_{R1} & R_{RR} \\ k_{R2} & R_{RR} \end{cases}$$

where k_{R1} and k_{R2} are the bit weights respectively for the long and short range scales in order to obtain r_{RR} in feet. When the Range Low Scale discrete is being received from the RR by the LGC, k_{R2} is used.

The range rate data obtained from the RR by the RR Data Read Routine is in the form of a binary data word S_{RR} which represents the count in the RR of a frequency comprising both the doppler frequency and a bias frequency (f_{BRR}) over a time interval τ_{RR}. At present, τ_{RR} is given as 80.000 milliseconds. To obtain the range rate ($\dot{r}_{RR}$) in feet per second, the following computation is made:

$$\dot{r}_{RR} = k_{RR} (S_{RR} - f_{BRR} \tau_{RR})$$

where k_{RR} is the scale factor required to obtain the range rate in feet per second and is of such a polarity as to make $\dot{r}_{RR}$ positive in the above equation for increasing range.

A summary of the processing constants required by the LGC for RR operation is given as follows:

f_{BRR}	Range rate bias frequency
τ_{RR}	Counting interval in RR for range rate measurements
k_{RR}	Scale factor to convert the range rate count obtained from the RR to feet per second for the counting interval τ_{RR}. The scale factor polarity is such as to make the converted result positive for increasing range.
k_{R1}	Bit weight in feet for long range scale.
k_{R2}	Bit weight in feet for short range scale.

5.2.4.2.2 Rendezvous Navigation Computations

During rendezvous phases RR navigation data are obtained by means of automatic rendezvous radar tracking of the CSM from the LM. These data are used to update the estimated six-dimensional state vector of either the LM or the CSM. The option of which state vector is to be updated by the RR tracking data is controlled by the astronaut as described in Section 5.2.1 and illustrated in Fig. 2.1-1. This decision will be based upon which vehicle's state vector is most accurately known, and upon which vehicle is performing the rendezvous. This process requires that the constant RR tracking angle biases be compensated for by estimating these biases along with the selected vehicle's state vector such that subsequent RR tracking angle data can be modified as shown in simplified form in Fig. 2.1-1.

This routine is used to process the CSM-tracking RR measurement data, and is used normally during lunar-orbit rendezvous in the lunar landing mission. The routine also can be used in earth orbit during alternate missions.

After the preferred LM attitude is achieved and RR tracking acquisition and lock-on is established (Section 5.2.4.1), the following tracking data are automatically acquired by the RR Data Read Routine (Section 5.2.4.2.1) at approximately one minute intervals:

Measured range, R_M

Measured range rate, $\dot{R}_M$

Measured shaft angle, β_M

Measured trunnion angle, θ_M

where the subscript M indicates the RR measured value. In addition to the above four measured quantities the time of the measurement and the three IMU gimbal angles are also recorded.

Although eight variables are estimated in the navigation procedure (six vehicle state-vector components and two RR angle biases), it is convenient to use the following nine-dimensional state vector:

$$\underline{x} = \begin{pmatrix} \underline{r} \\ \underline{v} \\ \delta_\beta \\ \delta_\theta \\ 0 \end{pmatrix} \qquad (2.4.1)$$

where $\underline{r}$ and $\underline{v}$ are the estimated position and velocity vectors, respectively, of the selected vehicle (LM or CSM) which is being updated, $\delta\beta$ and $\delta\theta$ are the estimates of the biases in the RR shaft and trunnion angles, respectively, and the ninth coordinate is a dummy variable. This type of RR tracking angle bias is referred to as a boresight bias and is one of two types which may be defined. With the LM attitude restriction mentioned in Section 5.2.1, either type of angle bias (tilt or boresight) can be used, and the boresight type indicated in Eq. (2.4.1) is most convenient.

Let $\underline{r}_L$, $\underline{v}_L$, $\underline{r}_C$ and $\underline{v}_C$ be the estimated position and velocity vectors of the LM and CSM, respectively, at the time of the measurement. Then, the measurement error variances, $\overline{\alpha^2}$, the nine-dimensional geometry vectors, $\underline{b}$, and the measured deviations, δQ, for the range and range rate measurements are computed as follows:

Measured range, R_M

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}(\underline{r}_{LC})$$

$$\overline{\alpha^2} = \text{maximum}(r_{LC}^2 \, \text{var}_R, \, \text{var}_{Rmin})$$

$$\underline{b}_0 = \mp \underline{u}_{LC} \tag{2.4.2}$$

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \underline{0}$$

$$\delta Q = R_M - r_{LC}$$

where var_R is the RR range error variance corresponding to a percentage error, and var_{Rmin} is the minimum RR range error variance.

Measured range rate, $\dot{R}_M$

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}(\underline{r}_{LC})$$

$$\underline{v}_{LC} = \underline{v}_C - \underline{v}_L \tag{2.4.3}$$

$$\dot{r} = \underline{v}_{LC} \cdot \underline{u}_{LC}$$

$$\overline{\alpha^2} = r_{LC}^2 \, \text{maximum}(\dot{r}^2 \text{var}_V, \, \text{var}_{Vmin})$$

$$\underline{b}_0 = \mp (\underline{u}_{LC} \times \underline{v}_{LC}) \times \underline{u}_{LC}$$

$$\underline{b}_1 = \mp \underline{r}_{LC}$$

$$\underline{b}_2 = \underline{0} \qquad (2.4.3 \text{ Continued})$$

$$\delta Q = r_{LC} (\dot{R}_M - \dot{r})$$

where var_V is the RR range-rate error variance corresponding to a percentage error, and var_{Vmin} is the minimum RR range-rate variance.

In Eqs. (2.4.2) and (2.4.3) the negative signs are selected if it is the LM state vector that is being updated, and the positive signs if it is the CSM state vector.

In order to process the RR angle data (β_M and θ_M), it is necessary to consider the relative orientations of the various coordinate systems. If $\underline{u}_X$, $\underline{u}_Y$ and $\underline{u}_Z$ are unit vectors along the X-, Y- and Z-axes, respectively, of the RR Measurement Coordinate System, then the measured shaft angle, β_M, and the measured trunnion angle, θ_M, are defined as shown in Fig. 2.4-7. In the figure, the vector $\underline{r}_{XZ}$ is the projection of the measured LM-to-CSM line-of-sight vector on the XZ-plane.

The RR Measurement Coordinate System is coincident with the Navigation Base Coordinate System since all RR performance specifications are referenced to the PGNCS navigation base. The unit vectors $\underline{u}_X$, $\underline{u}_Y$ and $\underline{u}_Z$ are then given in the Basic Reference Coordinate System by

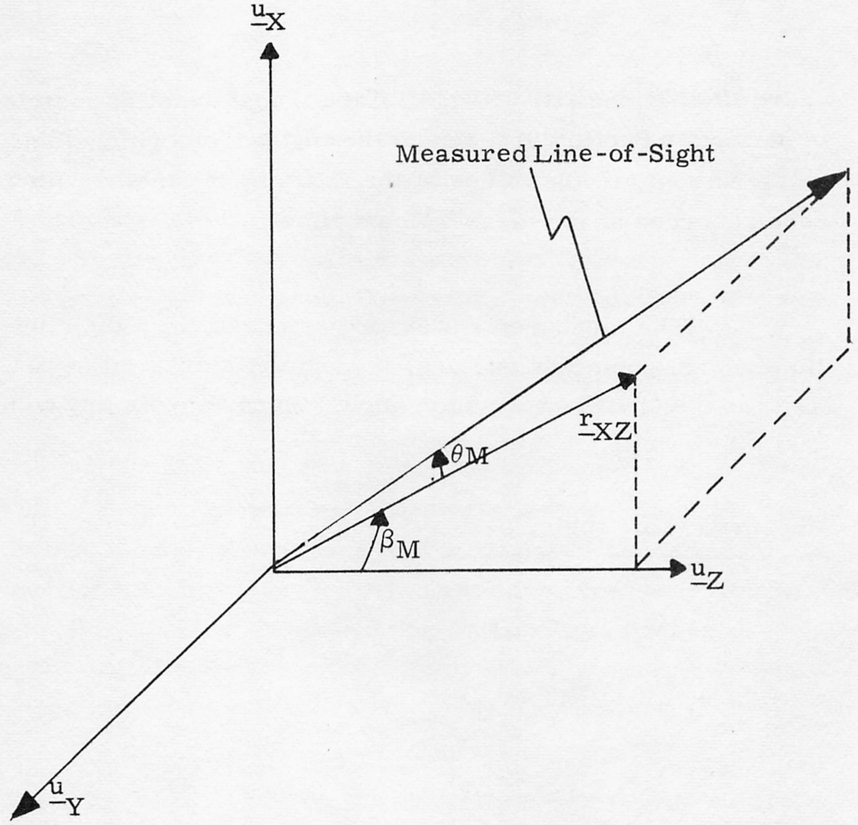


Figure 2.4-7 RR Measurement Coordinate System

$$\begin{pmatrix} \underline{u}_X^T \\ \underline{u}_Y^T \\ \underline{u}_Z^T \end{pmatrix} = [SMNB] \quad [REFSMMAT] \qquad (2.4.4)$$

where [SMNB] and [REFSMMAT] are transformation matrices as defined in Section 5.6.3 and the angles from which [SMNB] is determined are the values of the IMU gimbal angles which were recorded at the measurement time.

The measurement error variances, $\overline{\alpha^2}$, the nine-dimensional geometry vectors, $\underline{b}$, and the measured deviations, δQ, for the shaft and trunnion angle measurements are computed as follows:

<u>Measured shaft angle, β_M</u>

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}(\underline{r}_{LC})$$

$$S = -\underline{u}_{LC} \cdot \underline{u}_Y$$

$$r_{XZ} = r_{LC}\sqrt{1 - S^2}$$

$$\overline{\alpha^2} = \text{var}_\beta + \text{var}_{IMU} \qquad (2.4.5)$$

$$\underline{b}_0 = \mp \frac{1}{r_{XZ}} \text{UNIT}(\underline{u}_Y \times \underline{u}_{LC})$$

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}$$

$$\delta Q = \beta_M - \tan^{-1}\left(\frac{\underline{u}_X \cdot \underline{u}_{LC}}{\underline{u}_Z \cdot \underline{u}_{LC}}\right) - \delta\beta$$

where var_β is the RR shaft-angle error variance and var_{IMU} is the IMU angular error variance per IMU axis.

<u>Measured trunnion angle, θ_M</u>

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}\,(\underline{r}_{LC})$$

$$S = -\underline{u}_{LC} \cdot \underline{u}_Y$$

$$r_{XZ} = r_{LC}\sqrt{1 - S^2}$$

$$\overline{\alpha^2} = var_\theta + var_{IMU}$$

$$\underline{b}_0 = \mp \frac{1}{r_{XZ}} (\underline{u}_Y \times \underline{u}_{LC}) \times \underline{u}_{LC} \qquad (2.4.6)$$

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \begin{pmatrix} 0 \\ 1 \\ 0 \end{pmatrix}$$

$$\delta Q = \theta_M - \sin^{-1}(S) - \delta\theta$$

where var_θ is the RR trunnion-angle error variance.

In Eqs. (2.4.5) and (2.4.6) the negative signs are used if it is the LM state vector that is being updated, and the positive signs if it is the CSM state vector.

The data are incorporated into the state vector estimates by means of four calls to the Measurement Incorporation Routine (Section 5.2.3). The updated components of the nine-dimensional state vector resulting from each incorporation are used as initial conditions for the next update in the sequence.

Included in each use of the Measurement Incorporation Routine is the state vector update validity check, as described in Section 5.2.1 and illustrated in Fig. 2.4-8. Note that if δr and δv are displayed, a Source Code is also displayed indicating which of the four RR measurement parameters is responsible for the δr and δv. The Source Code is set to 1, 2, 3 and 4, respectively, for RR range, range rate, shaft angle and trunnion angle. In some cases, the identity of the RR measurement source is useful in indicating the type of corrective action required to remedy the situation.

The results of the processing of the RR measurement data are updated values of the estimated position and velocity vectors of the CSM or the LM and estimates of the RR angle biases. The two estimated vehicle state vectors are used to compute required rendezvous targeting parameters as described in Section 5.4.

For convenience of calculation in the LGC, Eqs. (2.4.5) and (2.4.6) are reformulated and regrouped as follows:

<u>Preliminary Radar Angle Calculation</u>

$$
\begin{aligned}
\underline{r}_{LC} &= \underline{r}_C - \underline{r}_L \\
\underline{u}_{LC} &= \text{UNIT}(\underline{r}_{LC}) \\
S &= -\underline{u}_{LC} \cdot \underline{u}_Y \\
r_{XZ} &= r_{LC}\sqrt{1 - S^2}
\end{aligned}
\qquad (2.4.7)
$$

<u>Measured shaft angle, β_M</u>

$$
\begin{aligned}
\overline{\alpha^2} &= r_{XZ}^2 \; (\text{var}_\beta + \text{var}_{IMU}) \\
\underline{b}_0 &= \mp \text{UNIT}(\underline{u}_Y \times \underline{u}_{LC}) \\
\underline{b}_1 &= \underline{0} \\
\underline{b}_2 &= \begin{pmatrix} r_{XZ} \\ 0 \\ 0 \end{pmatrix} \\
\delta Q &= r_{XZ}\left[\beta_M - \tan^{-1}\left(\frac{\underline{u}_X \cdot \underline{u}_{LC}}{\underline{u}_Z \cdot \underline{u}_{LC}}\right) - \delta\beta\right]
\end{aligned}
\qquad (2.4.8)
$$

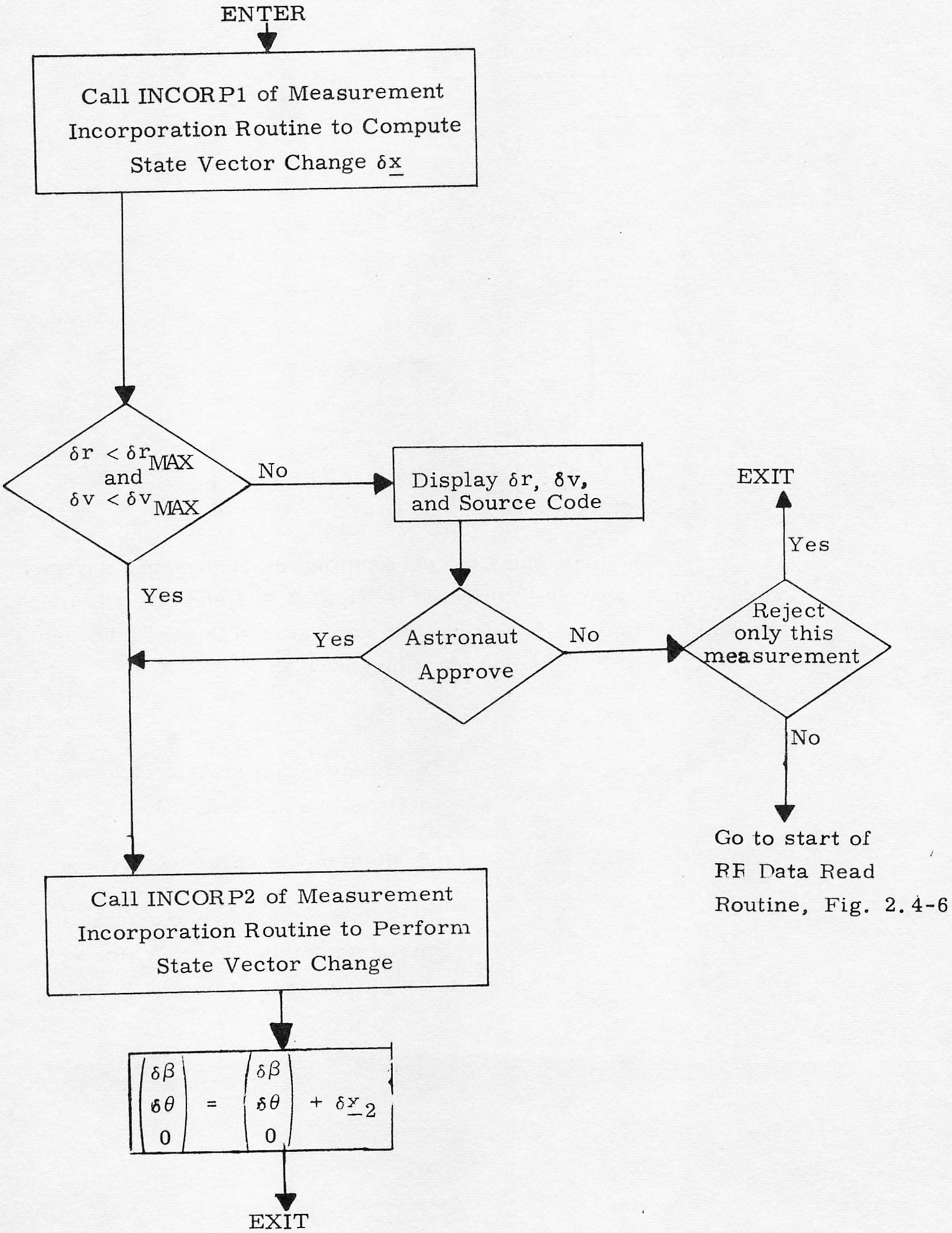


Figure 2.4-8 Rendezvous Navigation Measurement Incorporation Procedure

Measured trunnion angle, θ_M

$$\overline{\alpha^2} = r_{XZ}^2 (\mathrm{var}_\theta + \mathrm{var}_{IMU})$$

$$\underline{b}_0 = \mp (\underline{u}_Y \times \underline{u}_{LC}) \times \underline{u}_{LC}$$

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \begin{pmatrix} 0 \\ r_{XZ} \\ 0 \end{pmatrix} \qquad (2.4.9)$$

$$\delta Q = r_{XZ} \left[\theta_M - \sin^{-1} (S) - \delta\theta \right]$$

The procedure for performing the rendezvous navigation computations is illustrated in Figs. 2.4-9 and 2.4-10. It is assumed that the following items are stored in erasable memory at the start of the computation shown in Fig. 2.4-9:

$\underline{x}_C$	=	Estimated CSM state vector as defined in Section 5.2.2.6.
$\underline{x}_L$	=	Estimated LM state vector.
W	=	Six-dimensional error transition matrix associated with $\underline{x}_C$ or $\underline{x}_L$ as defined in Section 5.2.2.4.

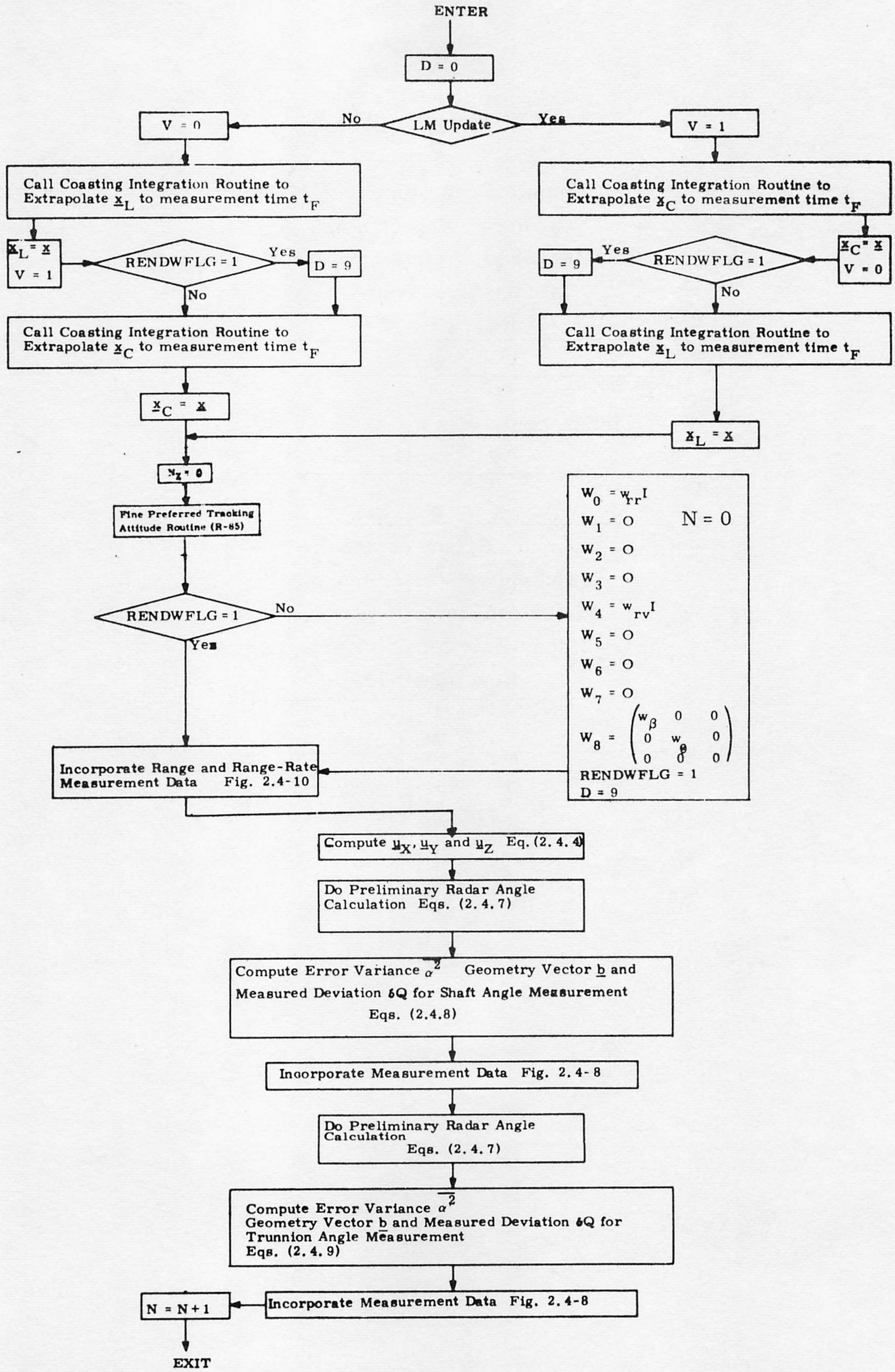


Figure 2.4-9 Rendezvous Navigation Computation Logic Diagram

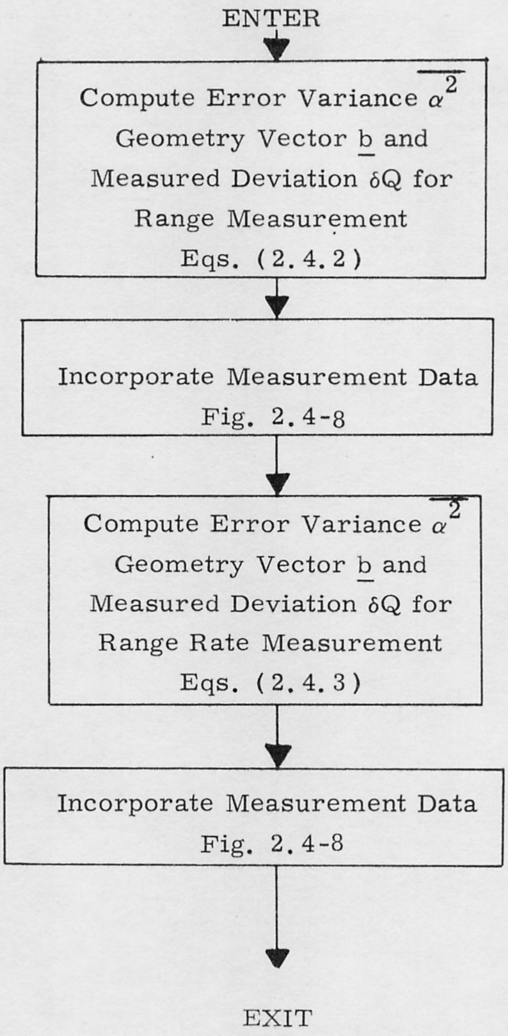


Fig. 2.4-10 RR Measured Range and Range Rate Incorporation

RENDWFLG = $\begin{cases} 1 \text{ for valid W-matrix} \\ 0 \text{ for invalid W-matrix} \end{cases}$

This flag or switch is maintained by programs external to the Rendezvous Navigation Routine. It indicates whether or not the existing W-matrix is valid for use in processing RR tracking data. The flag is set to zero after each of the following procedures:

1) State vector update from ground
2) Astronaut command
3) Overflow of W-matrix integration
4) When new W-matrix initialization values loaded by V67
5) Lunar ascent

$[REFSMMAT]$ = Transformation Matrix: Basic Reference Coordinate System to IMU Stable Member Coordinate System.

N = Number of sets of rendezvous navigation data already processed since the last maneuver, (set to 0 when W-matrix is initialized).

t_F = Measurement time.

$R_M, \dot{R}_M, \beta_M, \theta_M$ = RR measurement data

$w_{rr}, w_{rv}, w_\beta, w_\theta$ = Preselected W-matrix initial diagonal elements

Three IMU gimbal angles

Vehicle update mode

$$\text{SURFFLAG} = \begin{cases} 1 \text{ indicates RR angle measurements } (\beta_M \text{ and } \theta_M) \text{ are not to be used} \\ 0 \text{ indicates RR angle measurements } (\beta_M \text{ and } \theta_M) \text{ are to be used} \end{cases}$$

The term "SURFFLAG" is an abbreviated notation for the lunar surface flag; it is set to 1 by P-68 just after lunar landing and remains this value until launch. This flag indicates when the LM is on the lunar surface and is checked by a number of programs and routines.

The variables D and V are indicators which control the Coasting Integration Routine (Section 5.2.2) as described in Section 5.2.2.6, and I and O are the three-dimensional identity and zero matrices, respectively.

At the beginning of a new rendezvous sequence, the estimates of the RR angle biases, $\delta\beta$ and $\delta\theta$, may be initialized by the astronaut by direct addressing of erasable memory, and the W matrix is initialized by the program. During the remainder of the rendezvous sequence, $\delta\beta$ and $\delta\theta$ should not be re-initialized by the astronaut even though there may be additional W matrix initializations.

The RR-measurement-data incorporation procedure outlined above is repeated at approximately one-minute intervals throughout the rendezvous phase except during powered maneuvers. If the LM is the passive vehicle, the CSM rendezvous maneuvers are voice-linked to the LM as an ignition time and three velocity components in a CSM local vertical coordinate system, and then entered as updates to the estimated CSM state vector in the LGC. Upon receipt of these data, RR tracking and data processing should be suspended until after the maneuver. The update is accomplished by means of the Target Delta V Program, P-76 (Section 5.6.16). If the LM is the active vehicle, the estimated LM state vector is updated by means of the Average-G Routine (Section 5.3.2) during the maneuver providing a thrusting program has been performed; otherwise it is updated by the Impulsive Delta V Program (P77).

5.2.4.3 <u>RR Monitor Routine</u>

The logic associated with the RR Monitor Routine (R-25) is given in Fig. 2.4-11. This routine is initiated every 0.48 seconds by an automatic program interrupt and monitors various items such as the RR Auto Mode discrete, the RR CDU Fail discretes, and the angular excursions of the RR antenna.

Whenever the RR Auto Mode discrete changes status this routine resets various flags to insure proper initiation or termination of various radar control functions within the LGC. If the RR Auto Mode discrete has just been received from the RR as the result of placing the RR mode control switch in the LGC position, a turn-on sequence is initiated which zeroes the RR CDU's, determines the present RR antenna mode, and updates the Tracker Fail Light. The criterion used to determine the present RR antenna mode is the following:

$$\text{Mode 1: } 270^{\circ} \leq T \leq 90^{\circ}$$

$$\text{Mode 2: } 270^{\circ} > T > 90^{\circ}$$

where T is the RR antenna trunnion angle defined in Figs. 6.15-1 and 2.4-3. Note in Fig. 2.4-11 that the resulting antenna mode determination is indicated by the RR Antenna Mode Flag. When the RR Auto Mode discrete is removed by the RR, routine R-25 removes the RR Error Counter Enable from the RR CDU's to insure that no commands are being sent to the RR gyros.

If there is an RR CDU failure while the RR Auto Mode discrete is present, it is seen in Fig. 2.4-11 that the Tracker Fail Light is turned on. If P-20 or P-22 is in operation (i.e. the Rendezvous Flag is set), the Program Alarm Light is also turned on and an alarm code is stored.

After checking for RR CDU failure, it is seen in Fig. 2.4-11 that a number of conditions must be met before the routine will check to see if the RR antenna angles are within the tracking limits of the present mode (see Fig. 2.4-3). If all the conditions are met and the RR antenna is not within the present mode tracking limits, the routine will remove the RR Track Enable discrete from the RR, and cause the RR to be repositioned to the reference direction (see Fig. 2.4-3) for the present mode. During the repositioning of the RR this routine sets the Reposition flag to indicate to other programs and routines that this is taking place.

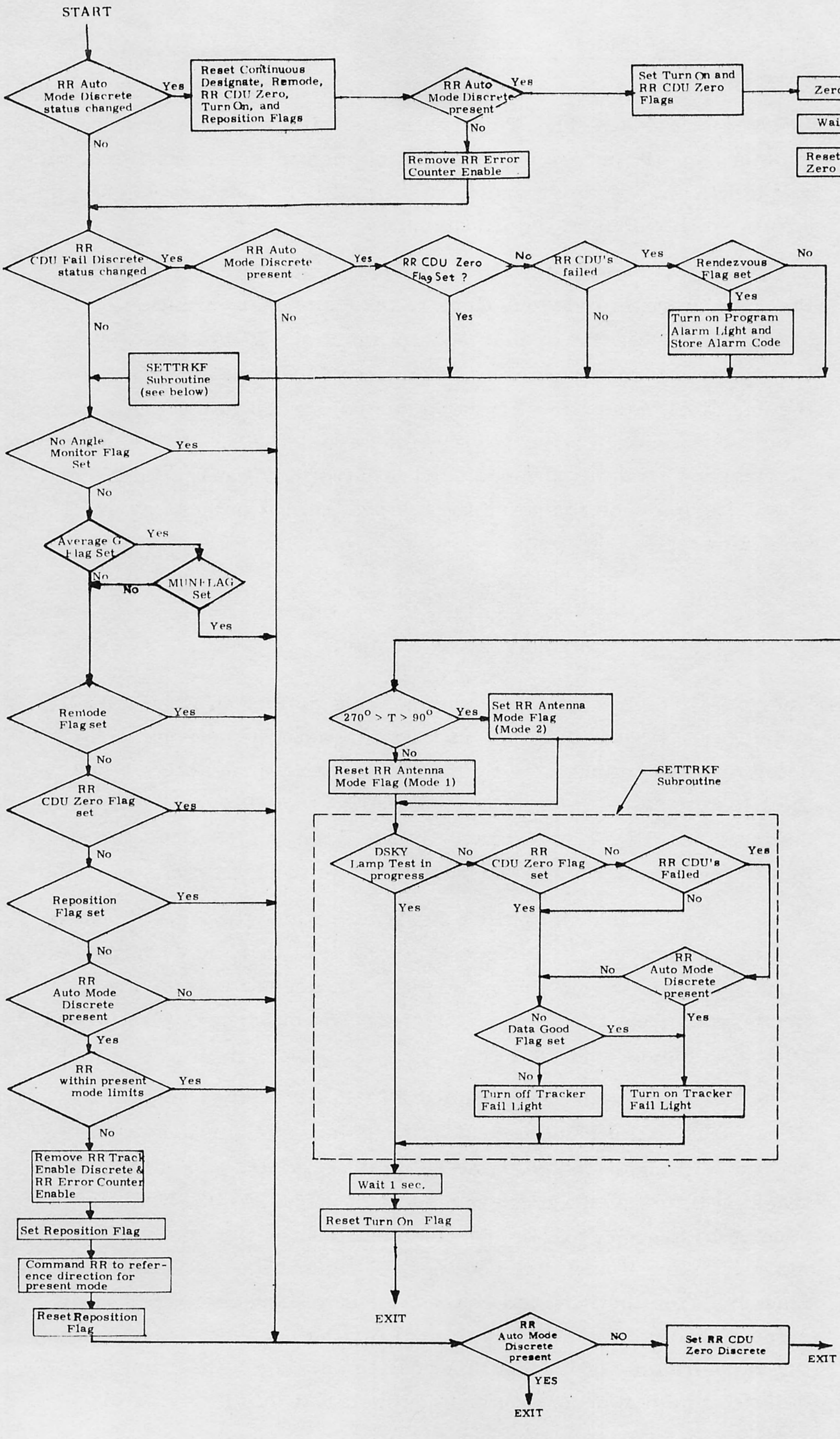


Figure 2.4-11 RR Monitor Routine

5.2.4.4 Preferred Tracking Attitude Routines

In the Rendezvous Navigation Program (P-20) use is made of two routines for aligning the LM +Z-axis with the LOS to the CSM. These routines are denoted as the Preferred Tracking Attitude Routine (R-61) and the Fine Preferred Tracking Attitude Routine (R-65). Both of these routines perform the alignment in the same manner except that routine R-65 is capable of repeating the alignment a specified number of times before returning to the program or routine which called it. Whenever continuous or fine Z-axis tracking is desired in program P-20, routine R-65 is called on a repetitive basis in order to stay within the minimum impulse limit cycle of the autopilot.

The logic associated with routines R-61 and R-65 is shown in Fig. 2.4-12 where it is seen that a check is made to insure that the Track flag is set before computing the unit vector $\underline{u}_{LOS}$ defining the LOS to the CSM in stable member coordinates and the angle ϕ between $\underline{u}_{LOS}$ in navigation base coordinates and the unit vector $\underline{z}_{NB}$ defining the +Z-axis of the Navigation Base Coordinate System. Using $\underline{u}_{LOS}$ and $\underline{z}_{NB}$ the Vecpoint Routine computes the new desired IMU gimbal angles (using the present desired IMU gimbal angles so as to prevent roll about desired vector). If the Attitude Control Switch is not in the Auto position, the desired IMU gimbal angles are converted to angular readings for display on the FDAI Ball by the method given in Section 5.6.12. If the Attitude Control Switch is in the Auto Position, it is seen in Fig. 2.4-12 that a check is made on the magnitude of ϕ with respect to 15 degrees to determine whether the vehicle attitude should be corrected by issuing the desired IMU gimbal angles to the RCS DAP or by using the Attitude Maneuver Routine (R-60 of Section 4).

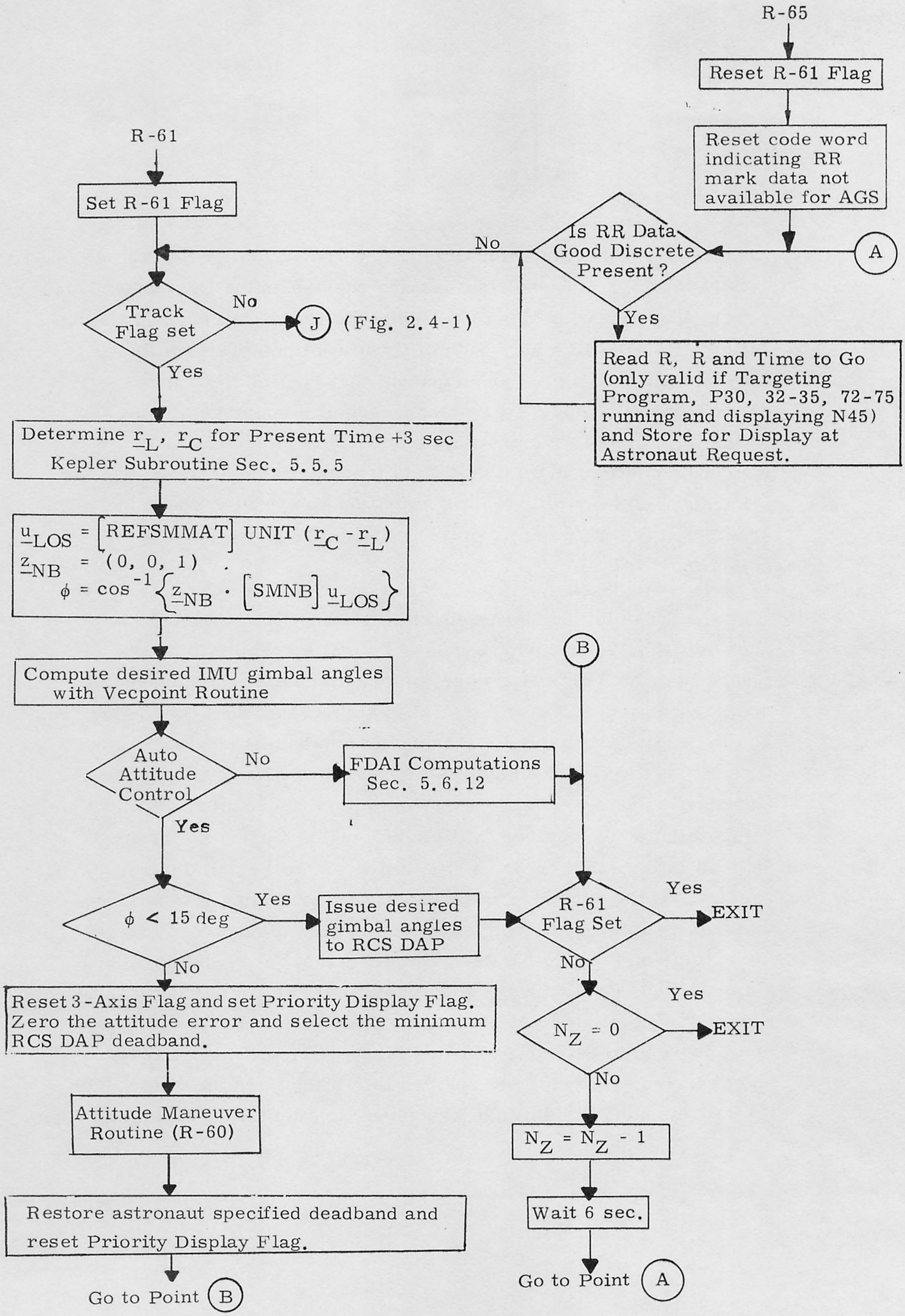


Fig. 2.4-12 Preferred Tracking Attitude Routines

In Fig. 2.4-12 it is seen that only one alignment is performed when routine R-61 is called. However, when routine R-65 is called, it is possible to have the alignment repeated N_Z times before returning to the program or routine which called routine R-65. When N_Z is greater than zero, the frequency of repetition of routine R-65 is approximately once every 8 seconds which is based upon a rough estimate of 2 seconds to perform the LOS and Vecpoint computations, and a 6 second wait period. The value of N_Z must be specified whenever routine R-65 is called.

In the target acquisition phase of program P-20 (see Section 5.2.4.1) only routine R-61 is used to align the LM +Z-axis to the LOS. However, during the operation of the RR Data Read Routine, routine R-65 is used to obtain continuous or fine Z-axis tracking. This is accomplished by automatically calling routine R-65 with $N_Z = 0$ near the middle of the state vector update computations of the Rendezvous Navigation Routine (see Fig. 2.4-9) and at the end of the RR Data Read Routine (see Fig. 2.4-6) with N_Z either equal to 0 or 5. During regular navigation updates with RR data the value of N_Z used at the end of the RR Data Read Routine is 5. A timeline depicting the operation of routine R-65 with N_Z set to 5 at the end of the RR Data Read Routine is shown in Fig. 2.4-13. Note that the frequency of reading the RR and updating the state vector is approximately once every 64 seconds. This frequency is based upon a rough estimate* of 22 seconds to read the RR, perform the state vector update computations, and use routine R-65 with $N_Z = 0$ in the Rendezvous Navigation Routine, and a rough estimate* of 42 seconds to use routine R-65 with $N_Z = 5$ at the end of the RR Data Read Routine.

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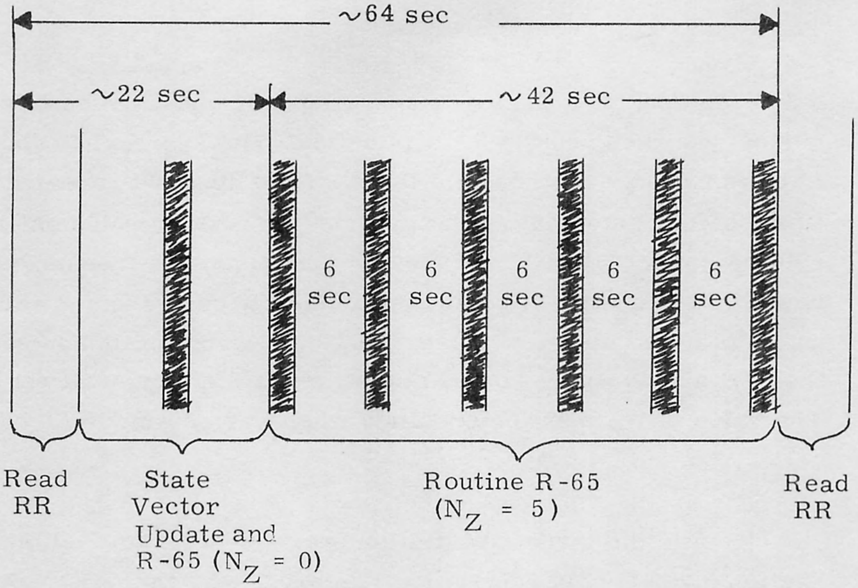
*The time to complete these calculations is highly dependent on other computational priorities. No explicit control of this procedure is exercised.



Note: Shaded intervals represent operation of Routine R-65 and are approximately 2 seconds duration each.

Fig. 2.4-13 Timeline for Fine LM +Z-Axis Tracking in Program P-20

In addition to program P-20, the Fine Preferred Tracking Attitude Routine (R-65) is also used to obtain continuous or fine Z-axis tracking in the Preferred Tracking Attitude Program (P-25 of Section 4). Program P-25 is used in place of program P-20 to keep the LM +Z-axis along the LOS to the CSM so that the CSM can optically track the LM optical beacon when the RR is not being used for navigation. This beacon is centered with respect to the LM +Z-axis and has a beamwidth of approximately 60 degrees. At the beginning of program P-25 the Track flag and the P25 Flag are set. Afterwards, routine R-65 is called with a specified value of 7 for N_Z, which causes routine R-65 to periodically perform the Z-axis alignment 8 times (see Fig. 2.4-12) before returning to program P25 where a check is made on the Track flag. If the Track flag has been reset by some other program, program P-25 will wait one minute, and then test the P-25 Flag. If the P-25 Flag is not set, the program exits. If it is set, the Track Flag is checked. If the Track Flag is not set, P-25 will wait one minute before repeating the flag checks described above. If the Track Flag is set, R-65 is called with N_Z set to 7.

5. 2. 5 RR LUNAR SURFACE NAVIGATION PROGRAM

5. 2. 5. 1 General Comments

The primary purpose of the RR Lunar Surface Navigation Program P-22 is to allow the LGC to update the CSM state vector using rendezvous radar (RR) tracking data prior to lunar launch in those cases in which a CSM state vector update cannot be obtained from the CMC or RTCC. This program essentially provides the LGC with a self contained capability of computing the parameters required for IMU alignment (P-57).

The RR Lunar Surface Navigation Program is normally used during at least one CSM orbital overpass prior to the intended launch orbit. A required astronaut input to program P-22 is the approximate expected lunar launch time. In normal operations the CSM performs a plane change maneuver at least two orbits prior to LM launch such that the CSM orbital plane contains the launch site vector at launch. In the case of loss of communications when P-22 is used to update the CSM state vector, the astronaut has an option to assume such a plane change maneuver has or has not been made. This decision is normally based on recorded block data for this phase of the mission. The program then directs the RR for initial acquisition and tracking. After RR lock-on and tracking have been established the navigation program uses range and range rate radar data to update the LGC estimate of the CSM orbital state vector. The operations mentioned above are described in more detail in the following sections.

5.2.5.2 CSM Orbital Plane Change Estimation Routine

As mentioned in Section 5.2.5.1, one of the purposes of this routine is to check that the CSM lunar orbit plane change maneuver has been accounted for in the estimated CSM state vector. In order to estimate the actual CSM maneuver, the estimated CSM orbital plane is rotated to contain the landing site at the launch time before the first RR data is used to update the CSM state vector. This procedure can be bypassed if the astronaut determines that the rotation should not be performed, as described in Section 5.2.5.1.

The geometry associated with the rotation is illustrated in Fig. 2.5-1 and the computation logic and the definitions of all variables are given in Fig. 2.5-2.

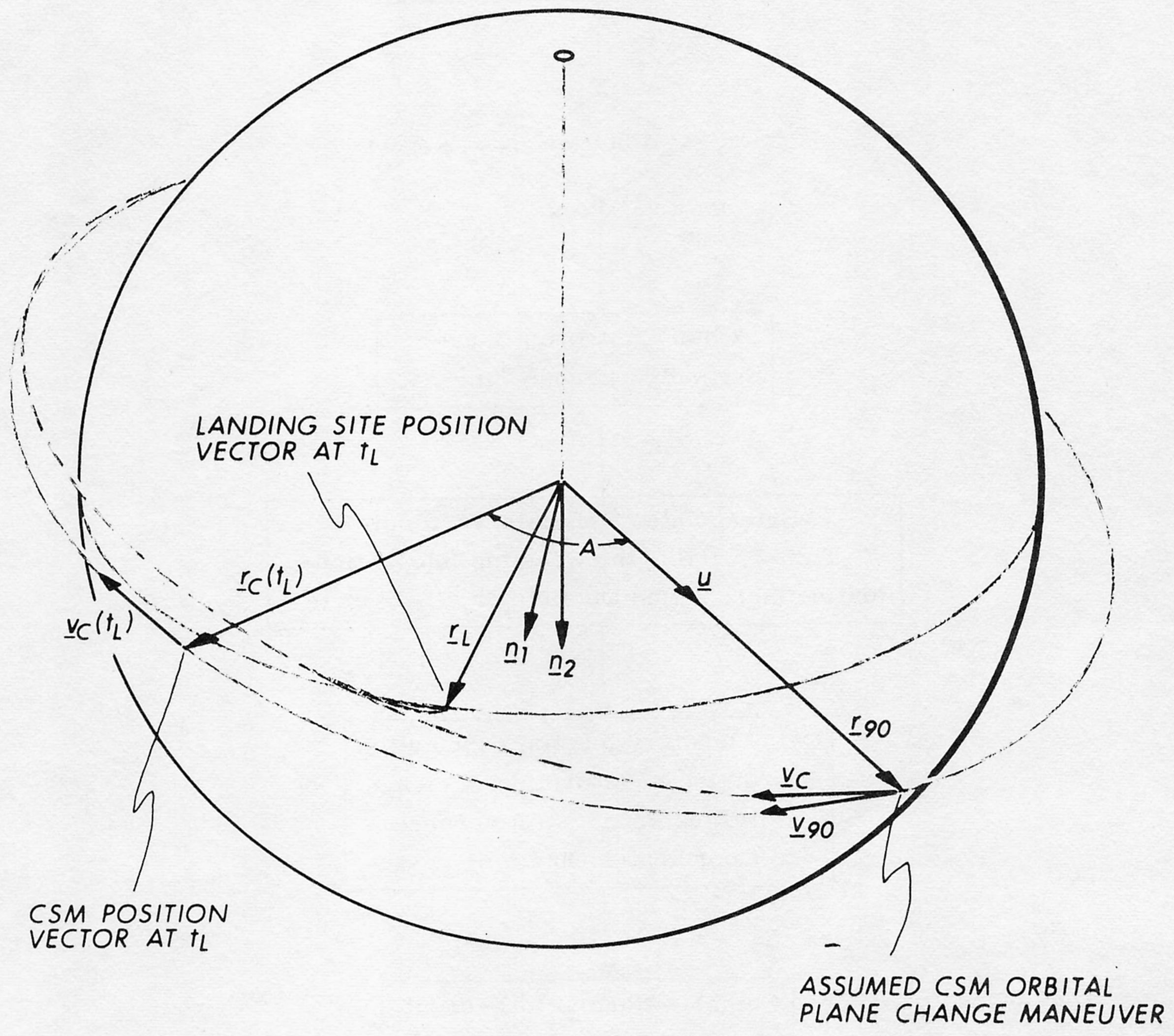


Fig. 2. 5-1 RR Lunar Surface Navigation CSM Orbital Plane Change Estimation

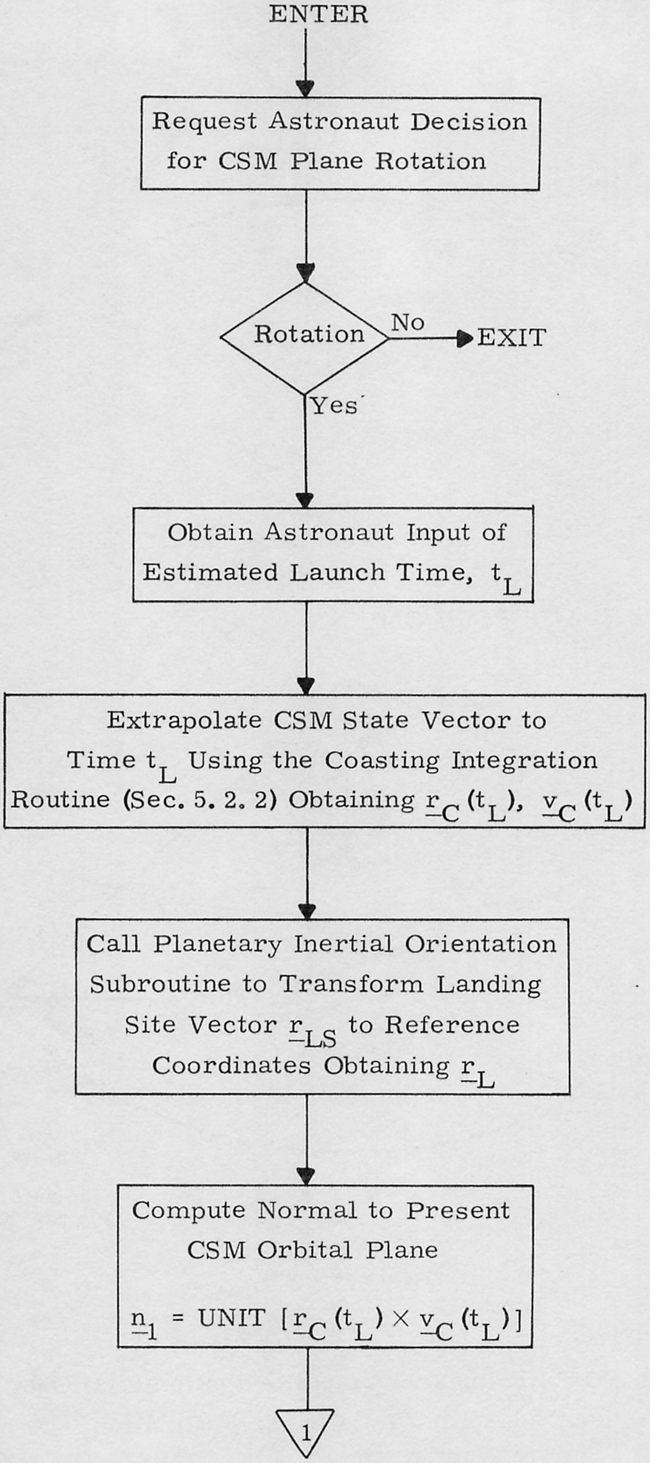


Figure 2.5-2 CSM Orbital Plane Change Estimation Routine Logic Diagram (page 1 of 2)

1

Determine Estimated CSM Orbital Plane Change Maneuver Location 90° Prior to Arrival at the Landing Site

$$\underline{u} = \text{UNIT}(\underline{r}_L \times \underline{n}_1)$$

↓

Compute the Angle A Between $\underline{u}$ and $\underline{r}_C(t_L)$

$$\cos A = \underline{u} \cdot \text{UNIT}[\underline{r}_C(t_L)]$$

$$\sin A = -\sqrt{1 - (\cos A)^2}$$

↓

Call Time-Theta Subroutine to Determine the CSM State Vector (t_{90}, $\underline{r}_{90}$, $\underline{v}_{90}$) at the Assumed Maneuver Point

↓

Compute the Normal to the CSM Orbital Plane Which Will Contain the Landing Site

$$\underline{n}_2 = \text{UNIT}(\underline{r}_{90} \times \underline{r}_L)$$

↓

Rotate $\underline{v}_{90}$ Into the Desired Orbital Plane

$$\underline{v}_C = v_{90}\, \text{UNIT}(\underline{n}_2 \times \underline{r}_{90})$$

$$\underline{r}_C = \underline{r}_{90}$$

$$t_C = t_{90}$$

↓

EXIT

Figure 2.5-2 CSM Orbital Plane Change Estimation Routine Logic Diagram (page 2 of 2)

5.2.5.3 Target Acquisition Routine

Since acquisition of the CSM with the LM Rendezvous Radar (RR) during lunar surface navigation is essentially the same as during rendezvous navigation, the same Acquisition Routine (see Fig. 2.4-1) is used for both except for slight differences in operation. Note in Fig. 2.4-1 that of the three modes (RR LGC, RR Manual, and RR Search) available for controlling the RR in target acquisition, only two (RR LGC and RR Search) are used in the Lunar Surface Navigation Program (P-22). In addition, the RR Search Mode can be selected only after the RR LGC Mode has failed to acquire the target.

The manner in which target acquisition is achieved with the RR LGC Mode is given in Figs. 2.4-2 and 2.5-3. Initially, steps are taken in Fig. 2.4-2 to insure that the RR antenna is in Mode 2. Afterward, the LOS to the CSM is computed and a check is made to see if it is within the angular tracking limits of Mode 2 (see Fig. 2.4-3). If the LOS is not within the Mode 2 tracking limits, the Lunar Surface RR Pre-Designate Routine (R-26) of Fig. 2.5-3 is used to determine whether the LOS will be inside the limits within approximately the next 10 minutes. This is accomplished by advancing the time t_{LOS} in 10-second increments, computing the LOS vector ($\underline{r}_{LOS}$) for each value of t_{LOS}, and checking to see if $\underline{r}_{LOS}$ is within the tracking limits of the present antenna mode (i.e., Mode 2). When a successful value of $\underline{r}_{LOS}$ is found, the routine advances t_{LOS} 10 more seconds, computes the corresponding $\underline{r}_{LOS}$, and starts designating the RR along $\underline{r}_{LOS}$ until the present time equals t_{LOS}. Subsequently, the RR Designate Routine in Fig. 2.4-2 is re-entered via P-22 and starts designating the RR toward the CSM. When the RR is being designated to the CSM, the LOS is periodically updated by updating the CSM position and velocity vectors ($\underline{r}_C$ and $\underline{v}_C$) with the Kepler Subroutine (Section 5.5.5). However, the LM position and velocity vectors ($\underline{r}_L$ and $\underline{v}_L$) used in the computation of the LOS are the vectors computed just before entering the RR Designate Routine rather than updated ones. The reason for not updating $\underline{r}_L$ and $\underline{v}_L$ during the designation is that updating is time consuming and results in very little improvement in LOS accuracy because of the small amount of lunar rotation during the operation of the RR Designate Routine. On the lunar surface, the LM position and velocity vectors ($\underline{r}_L$ and $\underline{v}_L$) are obtained by transforming the stored landing site position vector $\underline{r}_{LS}$ and the vector (0, 0, 1) from moon-fixed to basic reference coordinates with the Planetary Inertial Orientation Subroutine (Section 5.5.2) where $\underline{r}_L$ and $\underline{u}_Z$ are the respective vectors in basic reference coordinates and $\underline{v}_L$ is computed as follows:

$$\underline{v}_L = \omega_M \underline{u}_Z \times \underline{r}_L$$

where ω_M is the rotational rate of the moon with respect to inertial space.

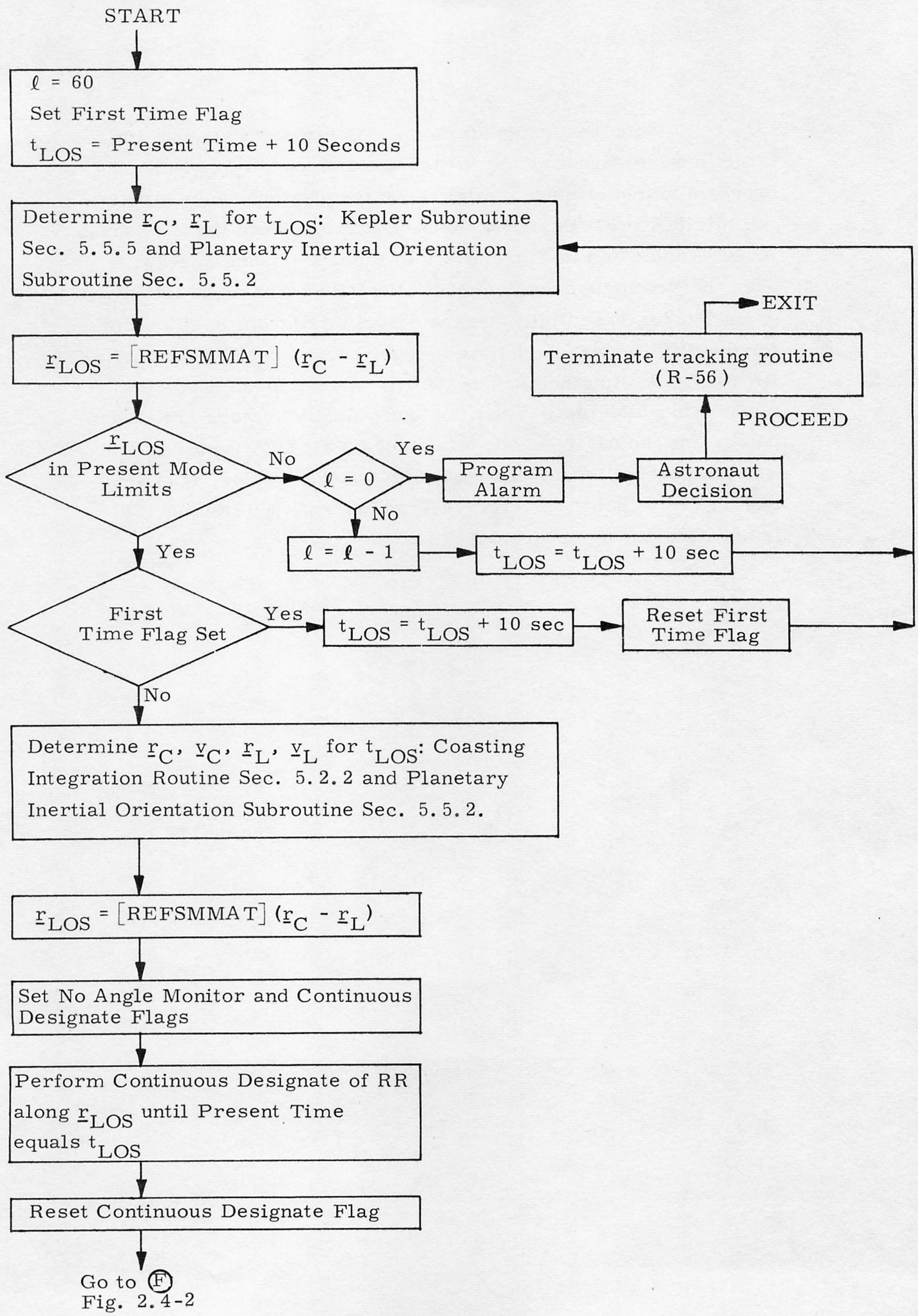


Fig. 2.5-3 Lunar Surface RR Pre-Designate Routine

If the RR Designate Routine is unable to acquire the target after designating the RR for a certain period of time, a program alarm is issued, whereupon the astronaut may either re-initiate the RR Designate Routine or proceed to the RR Search Mode as indicated in Fig. 2.4-2. In Figs. 2.4-1 and 2.4-5 it is seen that the logic associated with the RR Search Mode for program P-22 is essentially the same as for program P-20 except for the differences which have already been described for the RR Designate Routine such as the RR antenna mode used, the manner in which the LM position and velocity vectors are computed, and the nature of the test made on the LOS vector $\underline{r}_{LOS}$ with respect to the antenna mode limits. In addition, it is seen in Fig. 2.4-5 that the Preferred Tracking Attitude Routine is not used during program P-22.

5.2.5.4 Lunar Surface Navigation Routine

After target acquisition has been achieved with the Target Acquisition Routine (Section 5.2.5.3), the RR data is read by the RR Data Read Routine of Section 5.2.4.2.1 and used by the Lunar Surface Navigation Routine to update the navigation equations. In Fig. 2.4-6 it is seen that the operation of the RR Data Read Routine is essentially the same for both the Rendezvous and Lunar Surface Navigation Programs (P-20 and P-22) except that a 30^{o} check is not made on the direction of the RR during program P-22. The maximum frequency of update of the navigation equations in program P-22 is about once every 13 seconds. This frequency is based upon a rough estimate* of 11 seconds to read the RR and perform the navigation computations and a 2 second wait period before repeating the process. During the no-update mode in P-22 (i.e., when the No Update flag is set or the Update flag is not set), the frequency at which a complete set of RR data is read for downlink transmission is about 3.5 seconds -- 1.5 sec to read the RR data and perform the 3-deg test and a 2 sec wait period before repeating the process.

Although the RR Data Read Routine reads the RR angles along with the range and range rate, the angle data is not used for update purposes because of the uncertainties associated with the magnitude and nature of the RR angle biases which may be present during the large angular excursions of the RR with respect to the vehicle at this time. Thus, the estimated state vector in the Lunar Surface Navigation Routine is the six-dimensional CSM state vector, and only the range and range rate data (R_M and $\dot{R}_M$, respectively) are used in the navigation computations.

---

*The time to complete this calculation is highly dependent on other computational priorities. No explicit control of this procedure is exercised.

The computation logic for the Lunar Surface Navigation Routine is similar to the rendezvous navigation logic (Section 5.2.4.2) and is illustrated in Fig. 2.5-4. It is assumed that the following items are stored in erasable memory at the start of the procedure shown in the figure:

$\underline{x}_C$ = Estimated CSM state vector as defined in Section 5.2.2.6

W = Six-dimensional error transition matrix associated with $\underline{x}_C$ as defined in Section 5.2.2.4

$\underline{r}_{LS}$ = Estimated landing site or LM position vector on the surface of the moon in moon-fixed coordinates.

N = Number of measurement data points already processed.

$w_{\ell r}, w_{\ell v}$ = Preselected W-matrix initial diagonal elements.

$$\text{RENDWFLG} = \begin{cases} 1 \text{ for valid W-matrix} \\ 0 \text{ for invalid W-matrix} \end{cases}$$

This flag or switch is maintained by programs external to the Lunar Surface Navigation Routine. It indicates whether or not the W-matrix is valid for use in processing RR tracking data. The flag is set to zero after each of the following procedures:

1. State vector update from ground
2. Astronaut command
3. Overflow of W-matrix integration
4. New W-matrix initialization values are loaded via V67
5. Lunar ascent

The variables D and V are indicators which control the Coasting Integration Routine (Section 5.2.2) as described in Section 5.2.2.6, $\underline{u}_Z$ is a unit vector along the rotational axis of the moon, and ω_M is the lunar angular velocity.

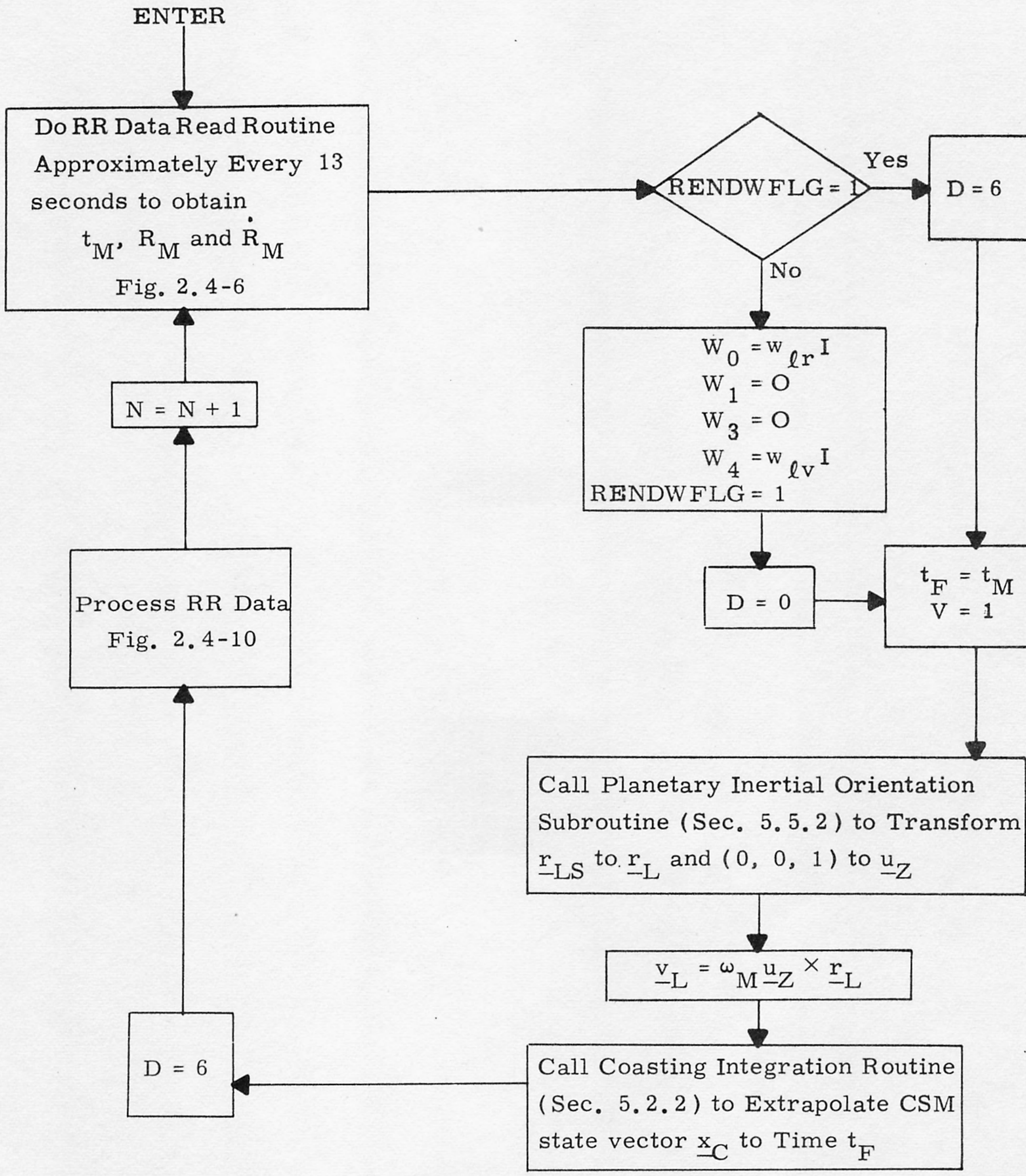


Fig. 2.5-4 Lunar Surface Navigation Routine Logic Diagram

5.3 POWERED FLIGHT NAVIGATION AND GUIDANCE

5.3.1 GENERAL COMMENTS

The objective of the powered flight guidance and navigation routines is to maintain an estimate of the LM state vector during thrusting maneuvers, and to control the thrust direction and duration such that the desired velocity cut-off conditions specified by the targeting routines of Section 5.4 are achieved. The powered flight navigation routine, used to maintain an estimate of the vehicle state vector during all thrusting conditions, is referred to as the Average-G Routine, and is presented in Section 5.3.2.

There are three basic powered flight guidance concepts used in the LGC for the Lunar Landing mission. The first is a velocity-to-be-gained concept with cross product steering (Section 5.3.3.4) that is used in each of the following two procedures:

1. Lambert Aim Point Maneuver Guidance (Section 5.3.3.5).

2. External ΔV Maneuver Guidance (Section 5.3.3.3.1).

These two procedures, based on the cross product steering concept, differ only in the unique generation of the desired velocity vector, $\underline{v}_R$.

The second basic guidance concept is the Quadratic Explicit Guidance of Section 5.3.4 which is used for the powered lunar landing maneuver. This concept is used to control the throttleable LM Descent Propulsion System (DPS) such that specified target or aim point conditions are achieved during the various phases of the landing maneuver. The landing maneuver target conditions for each phase are chosen to satisfy various DPS throttle and visibility constraints.

The third LGC guidance concept is the Linear Explicit Guidance of Section 5.3.5 used to control powered ascent maneuvers from the lunar surface to desired ascent injection conditions. This guidance concept is also used to control abort maneuvers initiated during the powered landing maneuver as described in Section 5.4.3.

5.3.2 POWERED FLIGHT NAVIGATION - AVERAGE-G ROUTINE

The purpose of the Powered Flight Navigation Subroutine is to compute the vehicle state vector during periods of powered flight steering. During such periods the effects of gravity and thrusting are taken into account. In order to achieve a short computation time the integration of the effects of gravity is achieved by simple averaging of the gravity acceleration vector. The effect of thrust acceleration is measured by the IMU Pulsed Integrating Pendulous Accelerometers (PIPA's) in the form of velocity increments ($\Delta \underline{v}$) over the computation time interval (Δt). The computations are, therefore, in terms of discrete increments of velocity rather than instantaneous accelerations. The repetitive computation cycle time Δt is set at 2 seconds to maintain accuracy and to be compatible with the basic powered flight cycle.

The Average-G Routine, in contrast to the Coasting Integration Routine, is used when a short computing time is required such as during powered flight. The Average-G Routine computations are illustrated in Figs. 3.2-1 and 3.2-2. In these figures the definitions of the following parameters are:

$\underline{r}(t)$	Vehicle position vector at time t.
$\underline{v}(t)$	Vehicle velocity vector at time t.
P_C	Planet designator $\begin{cases} 0 \text{ Earth} \\ 1 \text{ Moon} \end{cases}$
Δt	Computation cycle of 2 seconds.

$\Delta \underline{v}(\Delta t)$ The velocity vector sensed by the IMU PIPA's over the time interval Δt. This velocity vector increment is initially sensed in IMU or Stable Member Coordinates and then transformed to the Basic Reference Coordinate System when required.

$\underline{g}_p(t)$ Previous gravity acceleration vector at time t. This is a required initialization parameter and is supplied by the calling program.

$\underline{u}_r$ Unit vector in the direction of $\underline{r}$.

$\underline{u}_z$ Unit vector in the direction of the polar axis of the earth. = (0, 0, 1)

μ_E Earth gravitational constant.

μ_M Moon gravitational constant.

r_E Equatorial radius of the earth.

J Second-harmonic coefficient of the earth's potential function.

$\underline{g}_b(t)$ Component of the earth gravity acceleration vector representing earth oblateness effects.

With reference to Fig. 3.2-2 it can be seen that a single oblateness term is included in the earth gravity subroutine computation, but none for the lunar case.

The PIPA measured velocity $\Delta \underline{v}$ is compensated for instrument errors as described in Section 5.6.13 prior to being transformed into the Basic Reference Coordinate System and processed in the Average-G Subroutine of Fig. 3.2-1.

It should be noted that in the LGC there are two modes to the Average-G Routine during lunar orbit phases. In the first mode the Average-G Routine computations are referenced to the Basic Reference Coordinate System. The second mode is used in the lunar landing programs P-63, P-64 and P-66 and ascent and abort programs P-12, P-70 and P-71 in which the Average-G Routine computations are referenced to the Stable Member Coordinate System. In either mode the Average-G Routine output is finally referenced to the Basic Reference Coordinate System coordinates for telemetry down-link and extended verb computation purposes.

In addition, an up to date estimate of vehicle mass is made, similar to that used in descent (Section 5.3.4.3).

ENTER

$\underline{r}(t)$, $\underline{v}(t)$, Δt^{*}, $\Delta \underline{v}(\Delta t^{*})$, $\underline{g}_p(t)$, P_C

$$\underline{r}(t+\Delta t) = \underline{r}(t) + \Delta t^{*}\left[\underline{v}(t) + \underline{g}_p(t)\frac{\Delta t}{2} + \frac{\Delta \underline{v}(\Delta t^{*})}{2}\right]$$

Call Gravity Subroutine, (Fig. 3.2-2), $\underline{r}(t+\Delta t)$, P_C

Resume $\underline{g}(t+\Delta t)$, $\underline{g}_b(t+\Delta t)$

$$\underline{v}(t+\Delta t) = \underline{v}(t) + \frac{\Delta t}{2}\left[\underline{g}(t+\Delta t) + \underline{g}_p(t)\right] + \Delta \underline{v}(\Delta t^{*})$$

$$\underline{g}_p(t+\Delta t) = \underline{g}(t+\Delta t)$$

$$t = t + \Delta t$$

Return $\underline{r}(t)$, $\underline{v}(t)$, $\underline{g}_p(t)$, $\underline{g}_b(t)$

Figure 3.2-1 Average-G Subroutine

*This Δt is the actual computation cycle time rather than the nominal 2 sec.

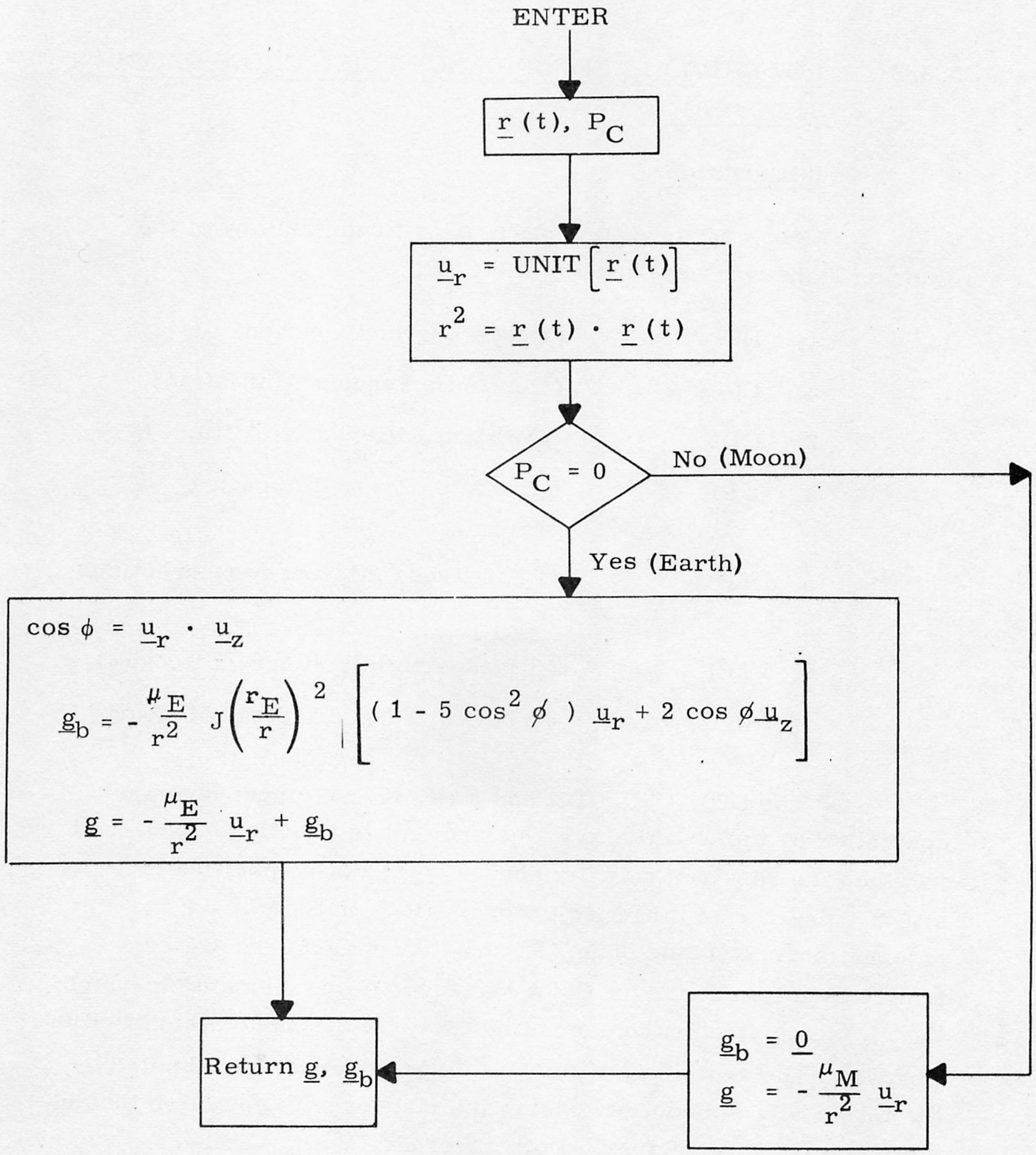


Figure 3.2-2 Gravity Subroutine

5.3.3 POWERED FLIGHT GUIDANCE USING CROSS PRODUCT STEERING

5.3.3.1 Introduction

Cross product steering is used to control any of the following maneuvers not done by the CSM.

1.	DOI	(Descent Orbit Injection)
2.	CSI	(Coelliptic Sequence Initiation)
3.	CDH	(Constant Differential Altitude)
4.	TPI	(Transfer Phase Initiation)
5.	TPM	(Rendezvous Midcourse Corrections)
6.	TEI	(Transearth Injection SPS Backup)
7.	External ΔV	(RTCC or CMC Targeted Maneuver)

The DOI, CSI, CDH and External ΔV maneuvers are controlled by the External ΔV Guidance Mode (Section 5.3.3.3.1) in which the required velocity change, $\underline{\Delta V}_{LV}$, is specified by either a pre-thrust targeting program (P32 or P33) or an external source (P30). The TPI and TPM maneuvers are controlled by a Lambert Aim Point Maneuver Guidance mode in which the $\underline{v}_R$ is computed each computing cycle by the ASTEER subroutine (for which the semi-major axis "a" is periodically recomputed by the LAMBERT subroutine) during the maneuver to establish the desired intercept trajectory. Both External ΔV and Lambert Aim Point guidance mode use the cross product steering concept in the LGC to control the thrust direction along the velocity-to-be-gained vector, and to terminate thrust when the desired velocity increment has been achieved.

Four subroutines are used repetitively in sequence (Section 5. 3. 3. 2) during cross product controlled maneuvers to accomplish this function. These are:

1. The Powered Flight Navigation Average-G Routine which computes the state vector accounting for the effects of thrust acceleration and gravity.

2. The Cross-Product Steering Subroutine which has 3 functions:

   a) generation of steering commands to the vehicle autopilot.

   b) computation of time-to-go before engine shut-off and the issuance of engine-off commands.

   c) incremental updating of the velocity-to-be-gained vector, (in the external ΔV mode only).

3. The Velocity-to-be-Gained Subroutine which repetitively solves the Lambert intercept problem when in the Lambert Aim Point guidance mode.

4. The ΔV Monitor Subroutine which detects engine ignition and unexpected thrust termination.

The Average-G Routine is described in Section 5. 3. 2. The other subroutines listed above are described in Sections 5. 3. 3. 4 to 5. 3. 3. 6. The Pre-Thrust Subroutines of Section 5. 3. 3. 3 initialize the powered maneuver programs for either the External ΔV or Lambert Aim Point guidance modes, and for the selected engine for the maneuver.

The LGC powered flight programs which use the Cross Product Steering Routine are:

P-40	Descent Propulsion System	(DPS)
P-41	Reaction Control System	(RCS)
P-42	Ascent Propulsion System	(APS)

Active steering and engine-off commands are provided by programs P-40 and P-42. Maneuvers using RCS translation control (P-41) are manually controlled and terminated by the astronaut while the LGC displays the required velocity-to-be-gained in spacecraft coordinates.

The functions of the External Delta V Pre-Thrust Program, P-30, are described in the pre-thrust subroutine description of Section 5.3.3.3.1.

5.3.3.2 Powered Flight Guidance Computation Sequencing

The time sequencing of the powered flight subroutines for External ΔV steering is shown in Figs. 3.3-1, thru 3.3-3 and for Lambert Aim Point steering in Figs. 3.3-4 thru 3.3-6.

Figures 3.3-1 and 3.3-4 show the sequencing during the ignition countdown which starts approximately 30 secs. before the nominal ignition time. Figures 3.3-2 and 3.3-5 show the normal sequencing for an engine-on time greater than 6 seconds as predicted by the pre-thrust subroutines. Figures 3.3-3 and 3.3-6 illustrate the sequencing during engine thrust termination.

The basic computation cycle time of the steering is 2 seconds and, as shown on the above figures, is initiated by the reading of the PIPA $\Delta \underline{v}$ registers. The various subroutines utilized during the 2 second cycle are sequenced in time as shown.

If the calculations conducted prior to the thrusting period (pre-thrust) indicate that the maneuver objective can be reached with a 6 second thrust period or less, provision is included (switches S_I and s_W) to preclude the generation of steering commands and engine commands during the thrusting period. In this case at the time of ignition the engine-off signal is set to be issued at the end of the estimated thrust period.

In addition to timing information, the sequence diagram of Figs. 3. 3-1 to 3. 3-6 also show the basic information utilized by each subroutine and its source.

The guidance computer program which controls the various subroutines to create a powered flight sequence is called the Servicer Routine . The sequence diagrams of Figs. 3. 3-1 to 3. 3-6 define what the Servicer Routine does, but do not show the logic details of how these functions are accomplished.

The subroutines listed in Figs. 3. 3-1 to 3. 3-6 are described in Section 5. 3. 2 and the following Sections 5. 3. 3. 3 to 5. 3. 3. 6. These sections should be referenced in tracing the powered flight computation sequencing.

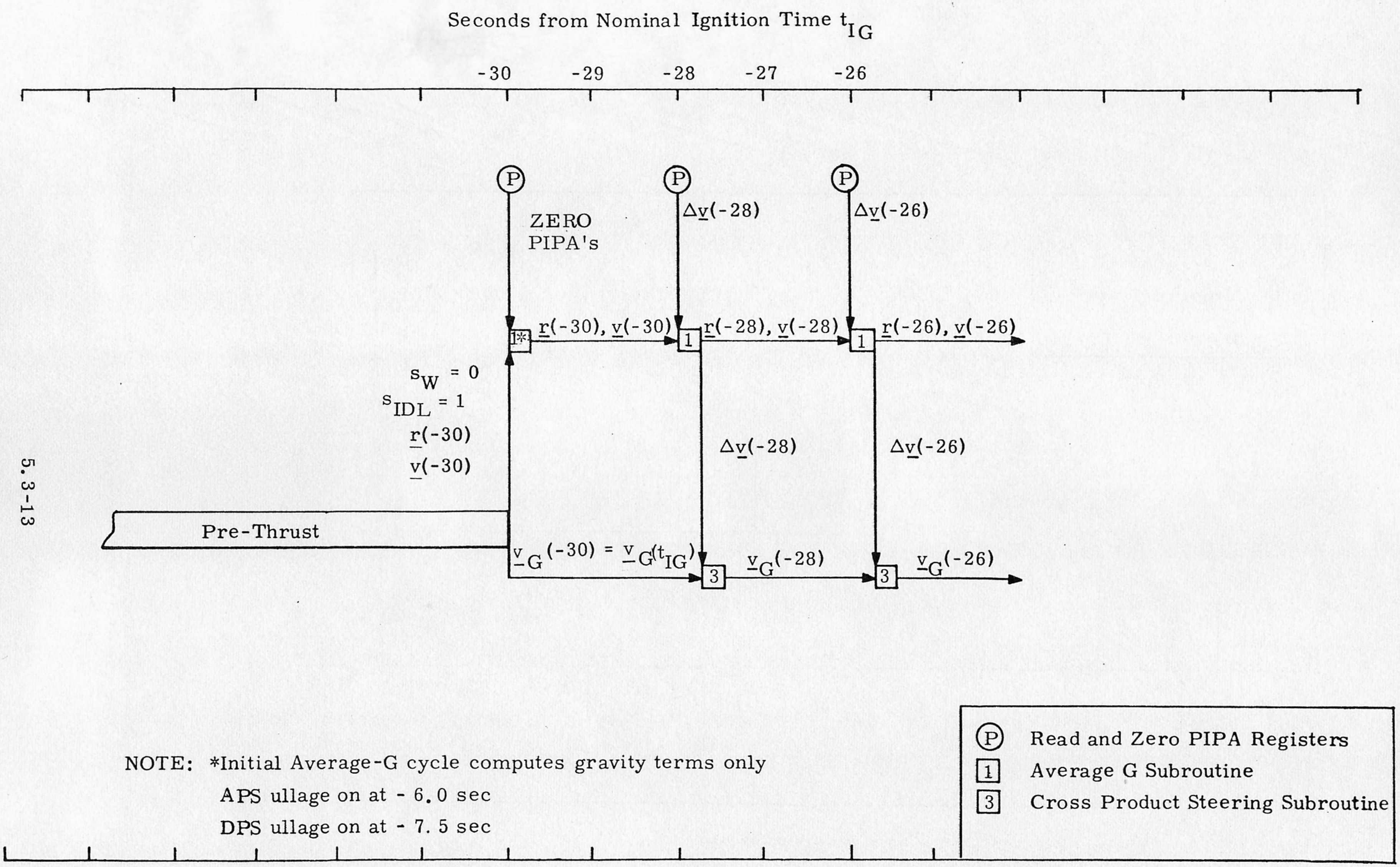


Fig. 3.3-1 Ignition Countdown - External ΔV Subroutine Sequencing for APS and DPS Maneuvers

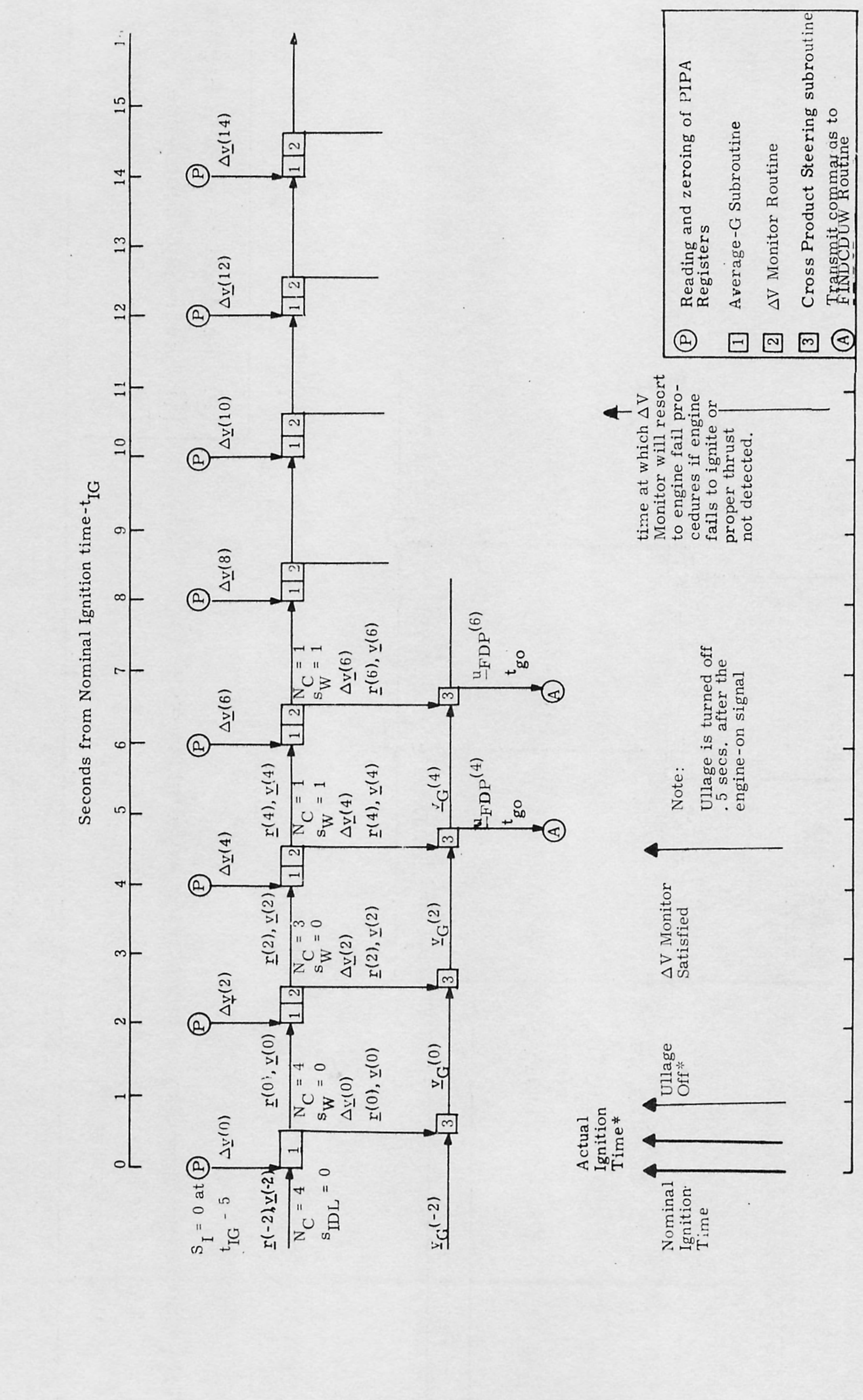


Figure 3.3-2. Normal External ΔV Subroutine Sequencing (Maneuver time greater than 6 sec) for APS and DPS

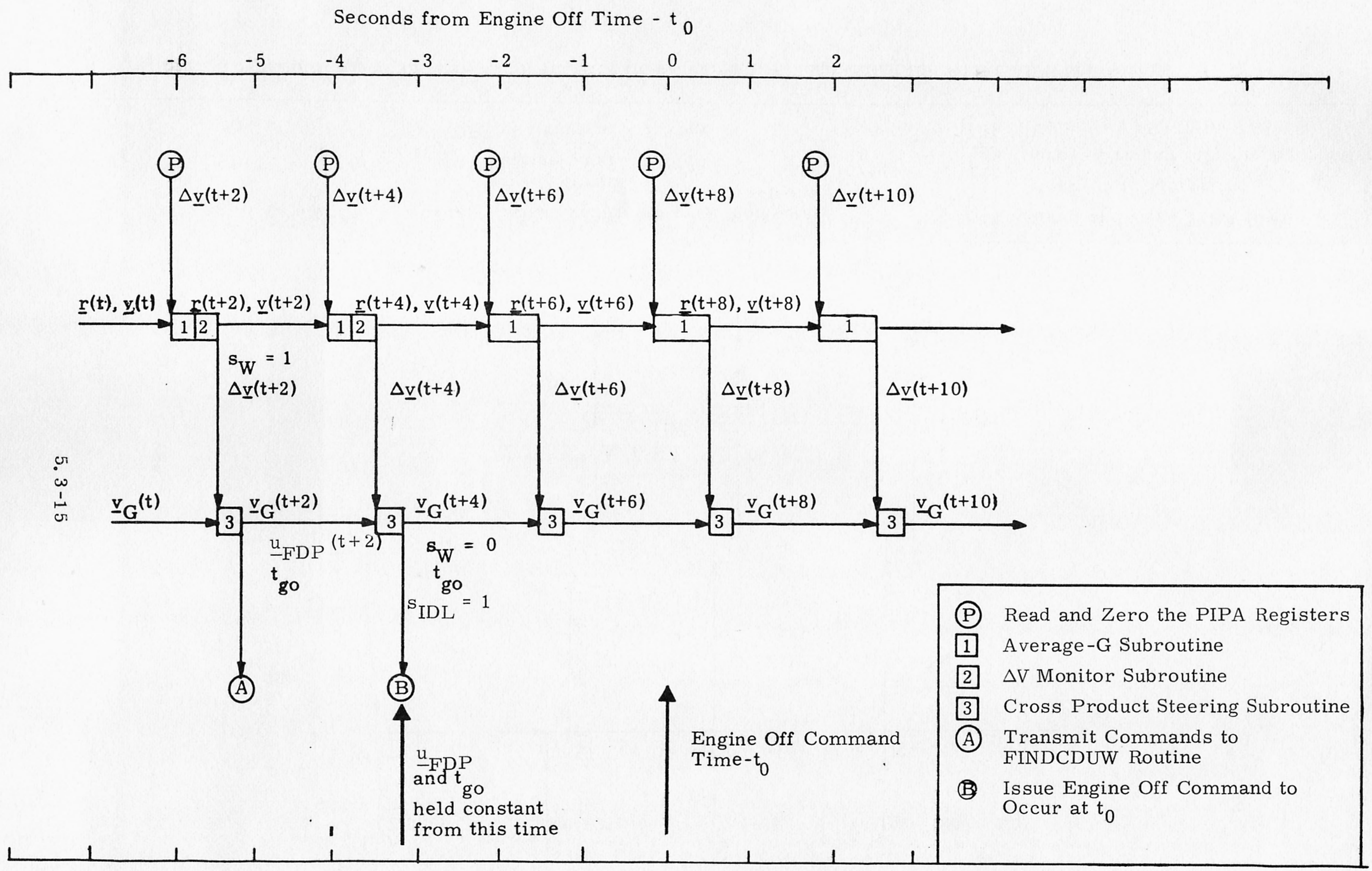


Fig. 3.3-3 Engine Off External ΔV Subroutine Sequencing for APS and DPS Maneuvers

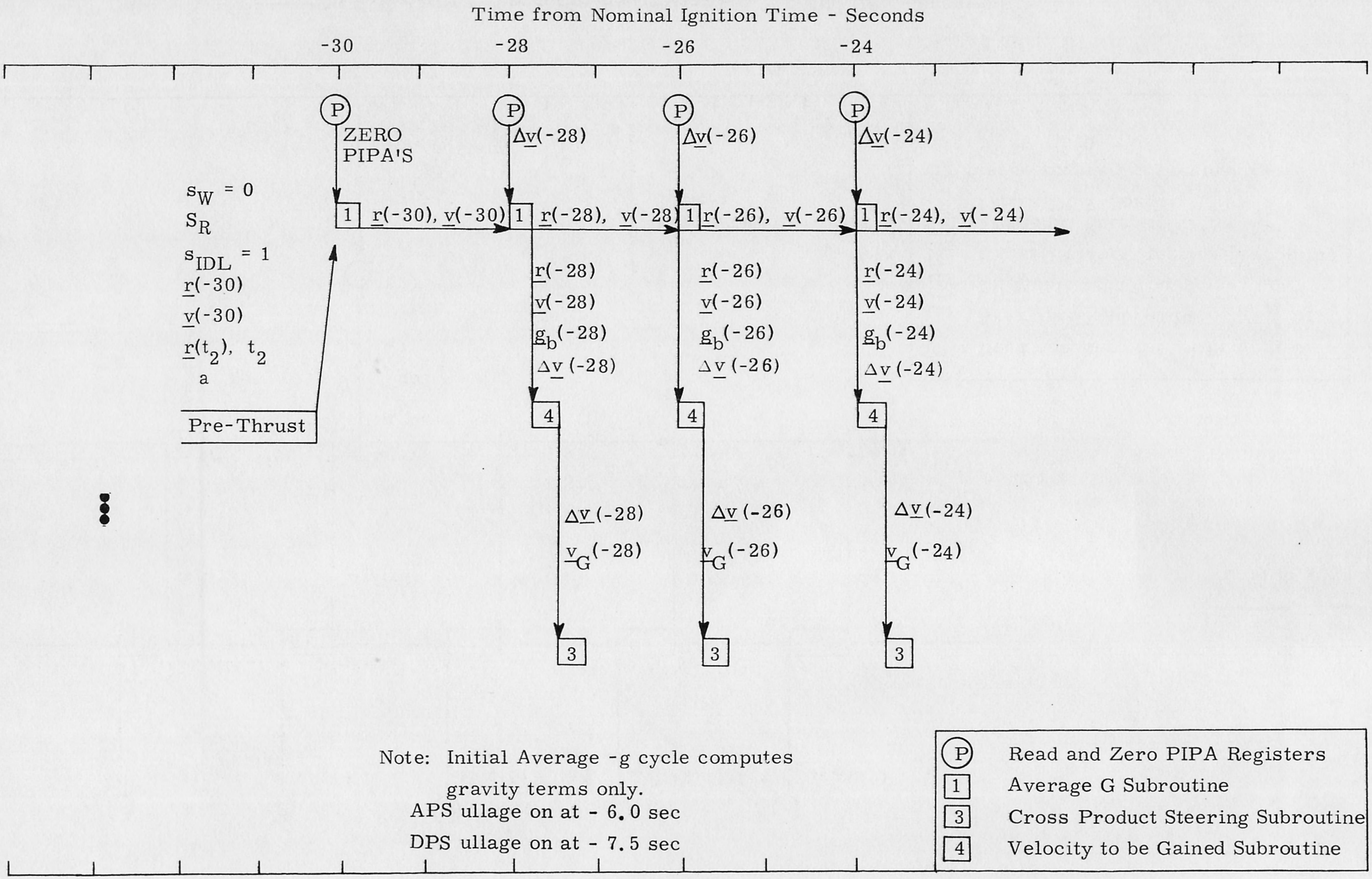


Fig. 3.3-4 Ignition Countdown - Lambert Aim Point Subroutine Sequencing for APS & DPS Maneuvers

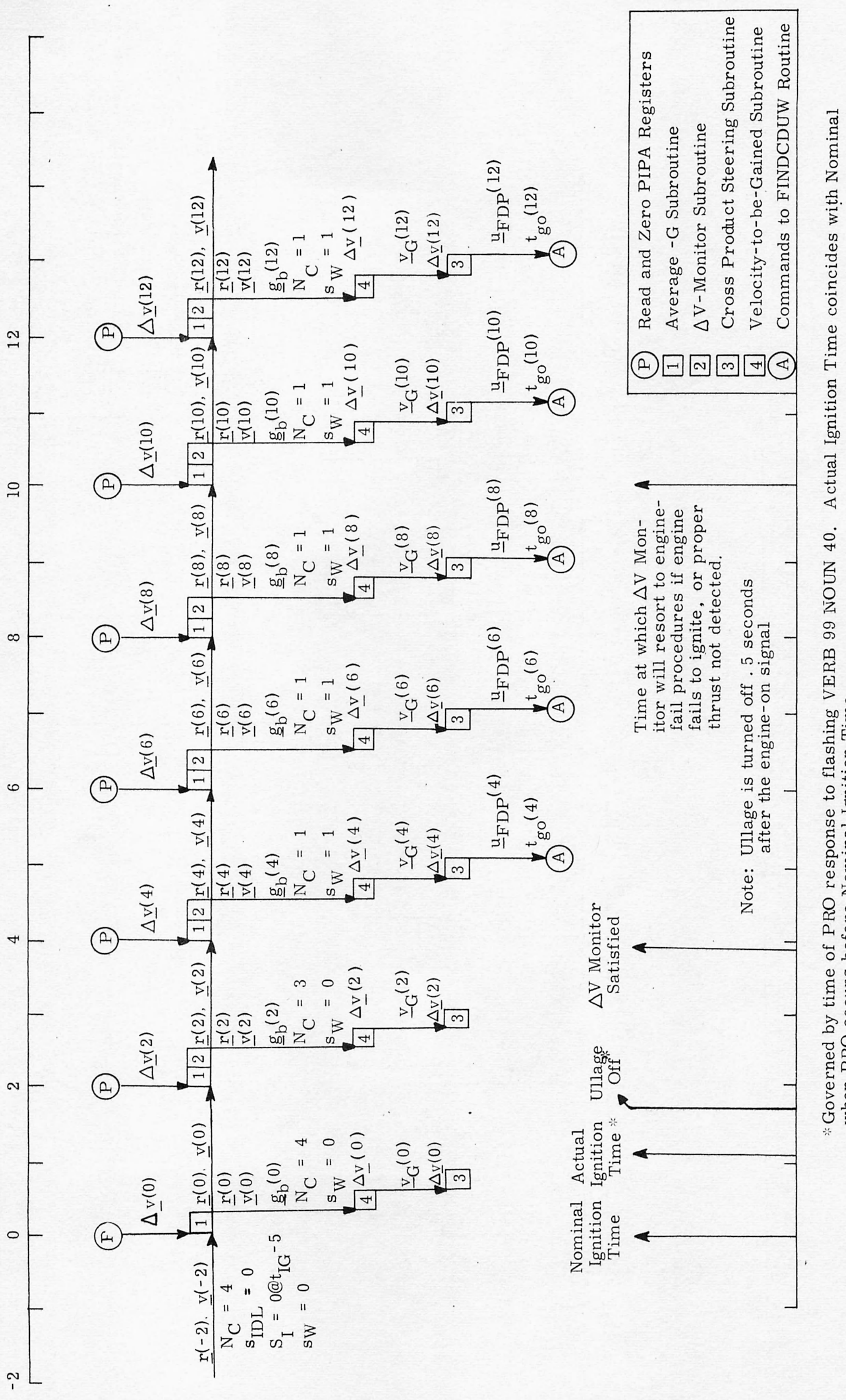


Figure 3.3-5. Normal Lambert Aim Point Subroutines Sequencing (Maneuver time greater than 6 sec) for APS and DPS

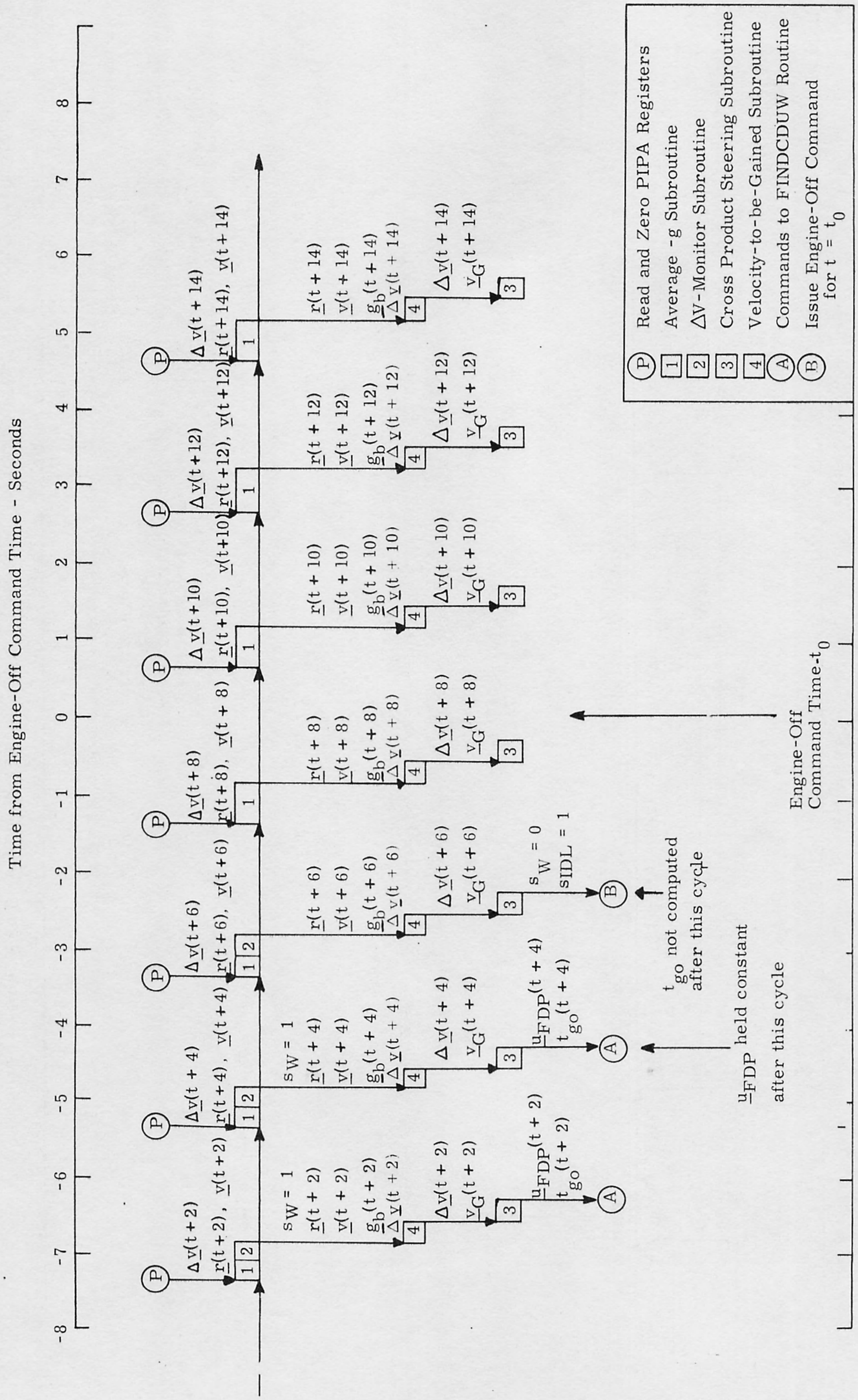


Figure 3.3-6. Engine-Off Lambert Aim Point Subroutine Sequencing

5.3.3.3 Pre-Thrust Computations

The objectives of the computations required prior to thrusting maneuvers are to determine the following factors:

1) The desired thrust direction at ignition.

2) The estimated duration of the powered maneuver to determine if there will be enough time to allow active steering, and in the case of the DPS if the throttle can be advanced to the full throttle position (FTP) after the 10% thrust start sequence.

3) Whether an IMU realignment is required to avoid gimbal lock.

4) Various parameters and variables required by subsequent powered flight routines.

The two major guidance modes using cross product steering are the External ΔV guidance mode and the Lambert Aim Point Maneuver guidance mode. The pre-thrust computations required for the External ΔV mode are presented in Section 5.3.3.3.1. Those required for the Lambert Aim Point Maneuver mode are described in Section 5.3.3.5.1. The initial IMU alignment computations and maneuver time logic is summarized in Section 5.3.3.3.3.

5.3.3.3.1 External ΔV Maneuver Pre-Thrust Computations

External ΔV maneuver guidance is normally used to control the CSI and CDH maneuvers of the concentric flight plan rendezvous profile, or an externally targeted maneuver in which a

constant thrust attitude is desired. The guidance program accepts input data via the DSKY (P-30) or the telemetry uplink (P-27) in the form of 3 components of an impulsive $\Delta \underline{V}_{LV}$ expressed in a local vertical coordinate system of the active vehicle at the ignition time t_{IG}. An approximate compensation for the finite maneuver time is made within the program by rotating the $\Delta \underline{V}_{LV}$ vector, and the guidance program issues commands to the spacecraft control system so as to apply the compensated velocity increment along an inertially fixed direction. The active vehicle state vector is normally either available or can be extrapolated to the ignition time in the LGC. If External ΔV guidance is used in cislunar space for midcourse corrections, the state vector must be uplinked for the ignition time along with the 3 components of the desired impulsive $\Delta \underline{V}_{LV}$.

The pre-thrust computations required for the External ΔV guidance mode are shown in Fig. 3.3-7. The following parameter definitions refer to this figure.

$\Delta \underline{V}_{LV}$ Specified velocity change in the local vertical coordinate system of the active vehicle at the time of ignition. This is an input parameter.

$$\Delta \underline{V}_{LV} = \begin{pmatrix} \Delta V_X \\ \Delta V_Y \\ \Delta V_Z \end{pmatrix}$$

t_{IG} Ignition time, an input parameter.

$\Delta \underline{V}_P$ The inplane velocity components of $\Delta \underline{V}_{LV}$ in the Basic Reference Coordinate System.

θ_T The approximate central angle traveled during the maneuver.

m Vehicle mass.

$\Delta \underline{V}$ Specified velocity change in Basic Reference Coordinates.

MGA Angle equivalent to the IMU middle gimbal angle when the vehicle X axis is aligned along $\Delta \underline{V}$. This angle is used to check for gimbal lock tolerance.

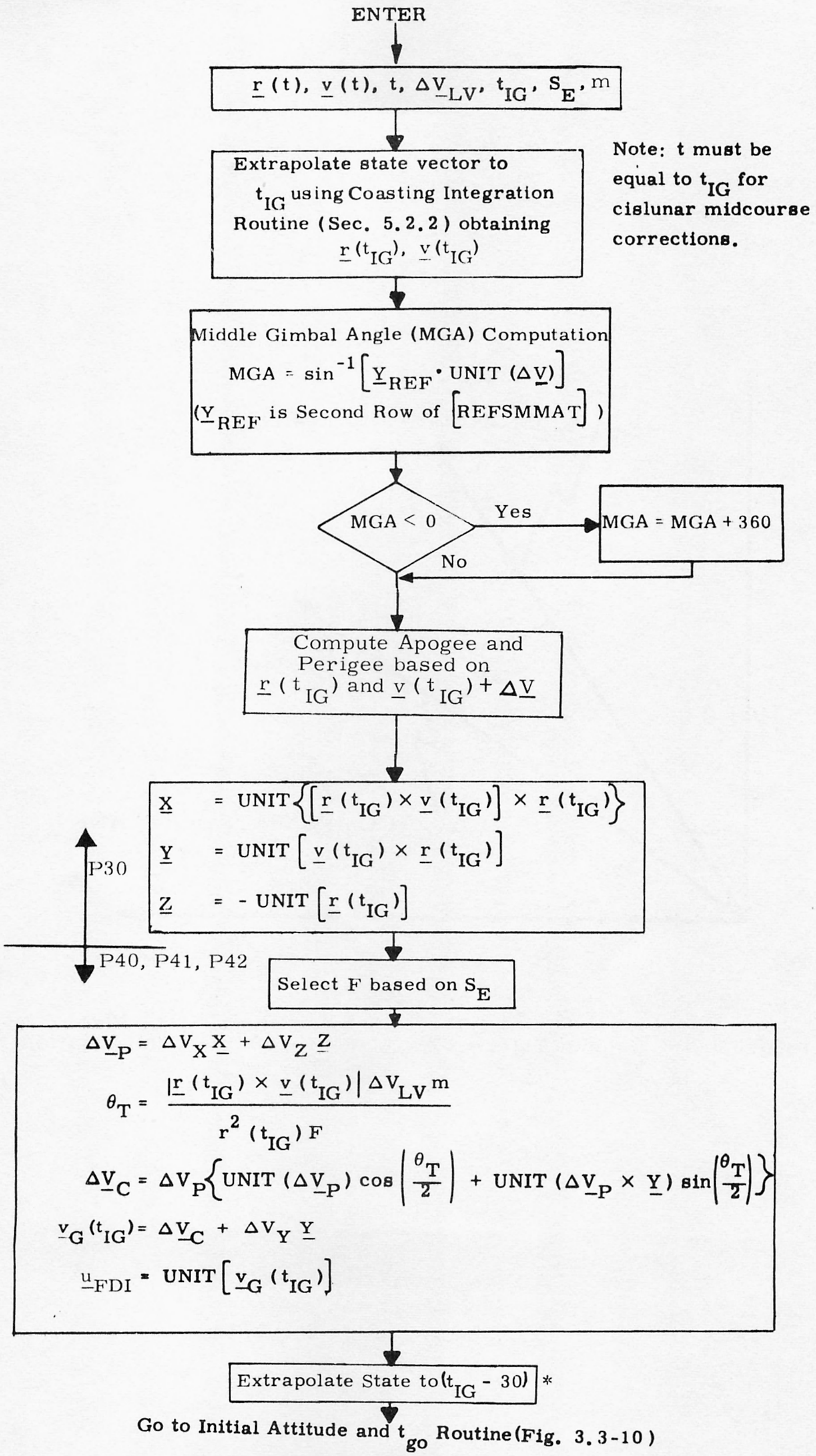


Fig. 3.3-7 External ΔV Prethrust Routine

*See Section 5.3.3.3.4 for ignition delay procedure

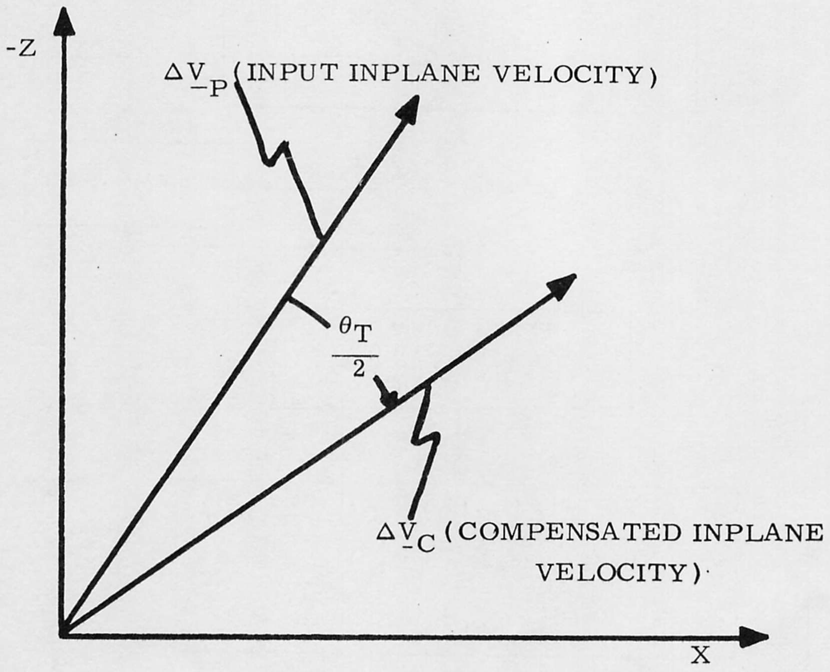


Figure 3.3-8 Inplane External ΔV Maneuver Angle Compensation

S_E	Engine Select Switch	$\begin{cases} -1 & \text{P-40 (DPS)} \\ 0 & \text{P-41 (RCS)} \\ 1 & \text{P-42 (APS)} \end{cases}$
F	Prestored nominal thrust based on the engine selection switch S_E F_{DPS} for the DPS (P-40) F_{APS} for the APS (P-42) F_{RCS} for the RCS (P-41) The values of the various engine thrust levels are listed in Sections 5.8 and 6. It should be noted that the DPS thrust level, F_{DPS}, is for the full throttle position (FTP) and the 26 second 10% thrust start sequence is not accounted for in the predicted maneuver time for the External ΔV guidance mode. In the case of P-41, RCS, the astronaut can select either a 2 or 4 jet translation maneuver.	
$\Delta \underline{V}_C$	The compensated inplane velocity-to-be-gained vector.	
$\underline{v}_G(t_{IG})$	Total velocity-to-be-gained at t_{IG}.	
$\underline{u}_{FDI}$	Unit vector in the desired initial thrust direction in the Basic Reference Coordinate System.	

The inplane External ΔV maneuver angle compensation involving $\Delta \underline{V}_P$ and $\Delta \underline{V}_C$ is illustrated in Fig. 3.3-8.

5.3.3.3.2 Lambert Aimpoint Maneuver Pre-Thrust Computations

As indicated in Fig. 3.3-4, the prethrust calculations of P40, P41 or P42 preceding t_{IG}-30 for a Lambert Maneuver provide a value of semi-major axis "a" for use by the ASTEER subroutine. Two cases of the generation of this parameter are pertinent: 1) when the central angle between the position vector at t_{IG} and the offset target vector (from a targeting program) is between 165° and 195° and 2) where this angle is less than 165° or greater than 195°.

In the former case the Lambert subroutine is called with the cone angle set to 45° ($S_R = 1$) and is recalled as frequently as possible to avoid computational difficulties associated with the 180° transfer case. In the latter case, the Lambert subroutine is called once with the cone angle set to 10° and is called subsequently to update "a" only on the basis of the time of the state vector used in the last computation of "a".

NOTE: Pages 5.3-25 and 5.3-26 deleted per PCR#823

5.3.3.3.3 Initial IMU Alignment and Maneuver Time-to-Go Computations

Both External ΔV and Lambert Aim Point guidance prethrust computations require the determination of the IMU middle gimbal angle, MGA, and estimated maneuver time, t_{go}, prior to engine ignition. These operations are shown in Fig. 3.3-7 and -10 with the exception that rendezvous MGA calculations are shown in Section 5.4. The input parameters required for these computations have been defined in Section 5.3.3.3.1. With reference to Fig. 3.3-10, the initial thrust direction unit vector, $\underline{u}_{FDI}$, and vehicle state vector are used to determine the Preferred IMU Alignment (Section 5.1.4.2). In Fig. 3.3-10 the following parameter definitions apply:

ΔV_K	Required ΔV Monitor Subroutine parameter (Sec. 5.3.3.6).
$\dot{m}$	Mass flow rate for APS.
$\Delta t_{Tail\text{-}off}$, V_e	Cross Product Steering Routine parameters (Sec. 5.3.3.4)
$\underline{X}_{SM}$, $\underline{Y}_{SM}$, $\underline{Z}_{SM}$	Unit vectors in Basic Reference Coordinates (BRC) of the directions of the IMU X, Y, and Z axes for preferred alignment.
N_C	Required ΔV Monitor Subroutine counter (Section 5.3.3.6).
[REFSMMAT]	Transformation matrix from the BRC System to the IMU or Stable Member Coordinate System (Section 5.6.3.4).
[SMNB]	Transformation matrix from the Stable Member Coordinate System to the Navigation Base Coordinate System (Section 5.6.3.2).
$\underline{u}_{FAB}$	Unit vector of the assumed thrust acceleration vector along the Navigation Base X axis.

In External ΔV guidance pre-thrust computations, the MGA display is computed as the angle between the input velocity vector $\Delta\underline{V}_{LV}$ transformed to the Basic Reference Coordinate System and $\underline{Y}_{REF}$ since $\underline{u}_{FDI}$ computations involving the inplane maneuver angle compensation cannot be made until the engine has been selected by the appropriate P40, 41 or 42 program.

ENTER

$\underline{u}_{FDI}$, $\underline{v}_G(t_{IG})$ m,

$$S_E \begin{cases} -1 & \text{P-40 (DPS)} \\ 0 & \text{P-41 (RCS)} \\ 1 & \text{P-42 (APS)} \end{cases}$$

Based on S_E set the following parameters:
ΔV_K, $\dot{m}$, $\Delta t_{Tail\text{-}off}$, V_e

Initial Vehicle Attitude and Preferred
IMU Alignment Computation

$$\underline{X}_{SM} = \underline{u}_{FDI}$$

$$\underline{Y}_{SM} = \begin{cases} \text{UNIT}\left[\underline{X}_{SM} \times \underline{r}(t_{IG})\right] & \text{If } \underline{r}(t_{IG}) \text{ not parallel to } \underline{u}_{FDI} \\ \text{UNIT}\left[\underline{X}_{SM} \times \underline{v}(t_{IG})\right] & \text{If } \underline{r}(t_{IG}) \text{ parallel to } \underline{u}_{FDI} \end{cases}$$

$$\underline{Z}_{SM} = \text{UNIT}\left[\underline{X}_{SM} \times \underline{Y}_{SM}\right]$$

$$s_W = 0$$

$$N_C = 4$$

$$\underline{u}_{FAB} = \begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}$$

$$\underline{u}_{FDB} = \left[\text{SMNB}\right]\left[\text{REFSMMAT}\right]\underline{u}_{FDI}$$

Maneuver vehicle attitude using the Vecpoint mode of the Kalcmanu Routine (Sec. 3) such that the resulting [SMNB] makes $\underline{u}_{FDB}$ equal to $\underline{u}_{FAB}$. Astronaut may choose to align IMU using P52 when predicted gimbal angles for final attitude are displayed.

1

Figure 3.3-10 Initial Attitude and t_{go} Routine
(Page 1 of 2)

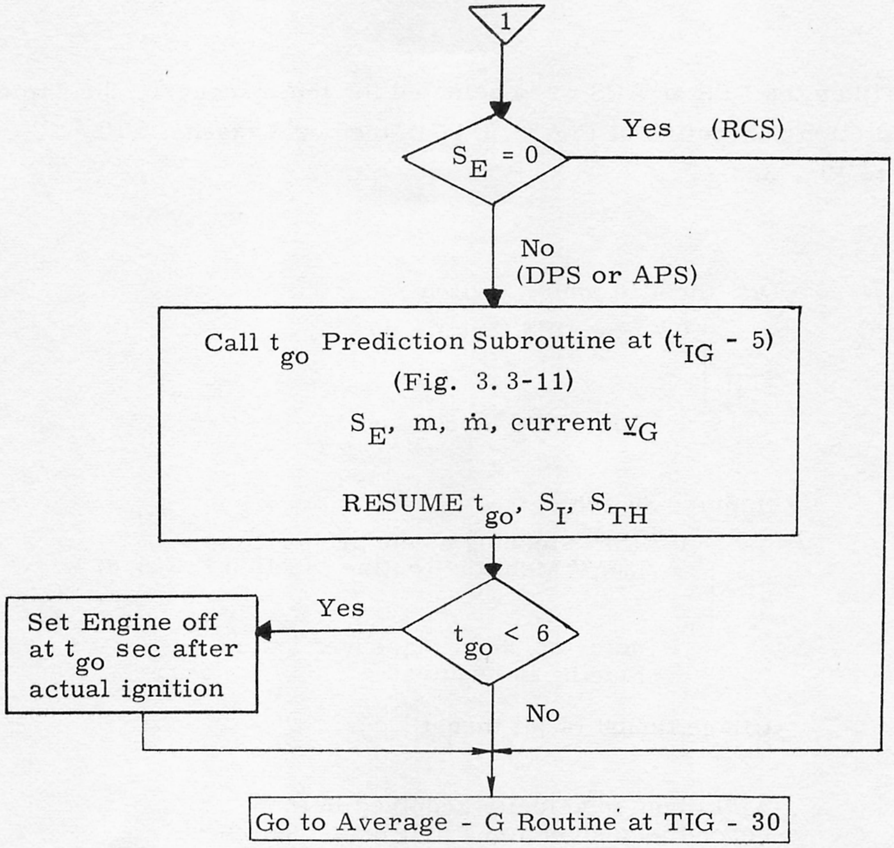


Figure 3.3-10 Initial Attitude and t_{go} Routine (Page 2 of 2)

If either the DPS or APS were selected for the maneuver, the Time-to-Go Prediction Subroutine of Fig. 3.3-11 is then processed. With reference to Fig. 3.3-11:

S_{TH} DPS throttle inhibit switch

$$S_{TH} \begin{cases} 0 & \text{Allow DPS throttle up} \\ 1 & \text{Inhibit DPS throttle up} \end{cases}$$

S_I Impulse Switch

$$S_I \begin{cases} 0 & \text{Allow steering at the proper time via } \Delta V \text{ Monitor Routine (Section 5.3.3.6)} \\ 1 & \text{Indicates short maneuver and no steering is required} \end{cases}$$

F_L Ullage thrust (4 jet thrust)

K_1 APS impulse velocity acquired in a one second maneuver for a unit mass vehicle

K_2 APS minimum impulse constant

K_3 APS minimum impulse constant equal to the slope of minimum impulse curve.

K_4 10% DPS engine thrust

F_{APS} APS Thrust

The initial computation in Fig. 3.3-11 estimates the velocity-to-be-gained after 6.5 seconds of ullage. If the DPS were selected, (P-40), the maneuver time, t_{go}, is then computed on the basis of 10% thrust. If this time is less than 6 seconds no active guidance steering is attempted ($S_I = 1$) and the vehicle attitude is maintained at the pre-thrust alignment throughout the maneuver. If the DPS t_{go} is greater than 95 seconds normal steering will

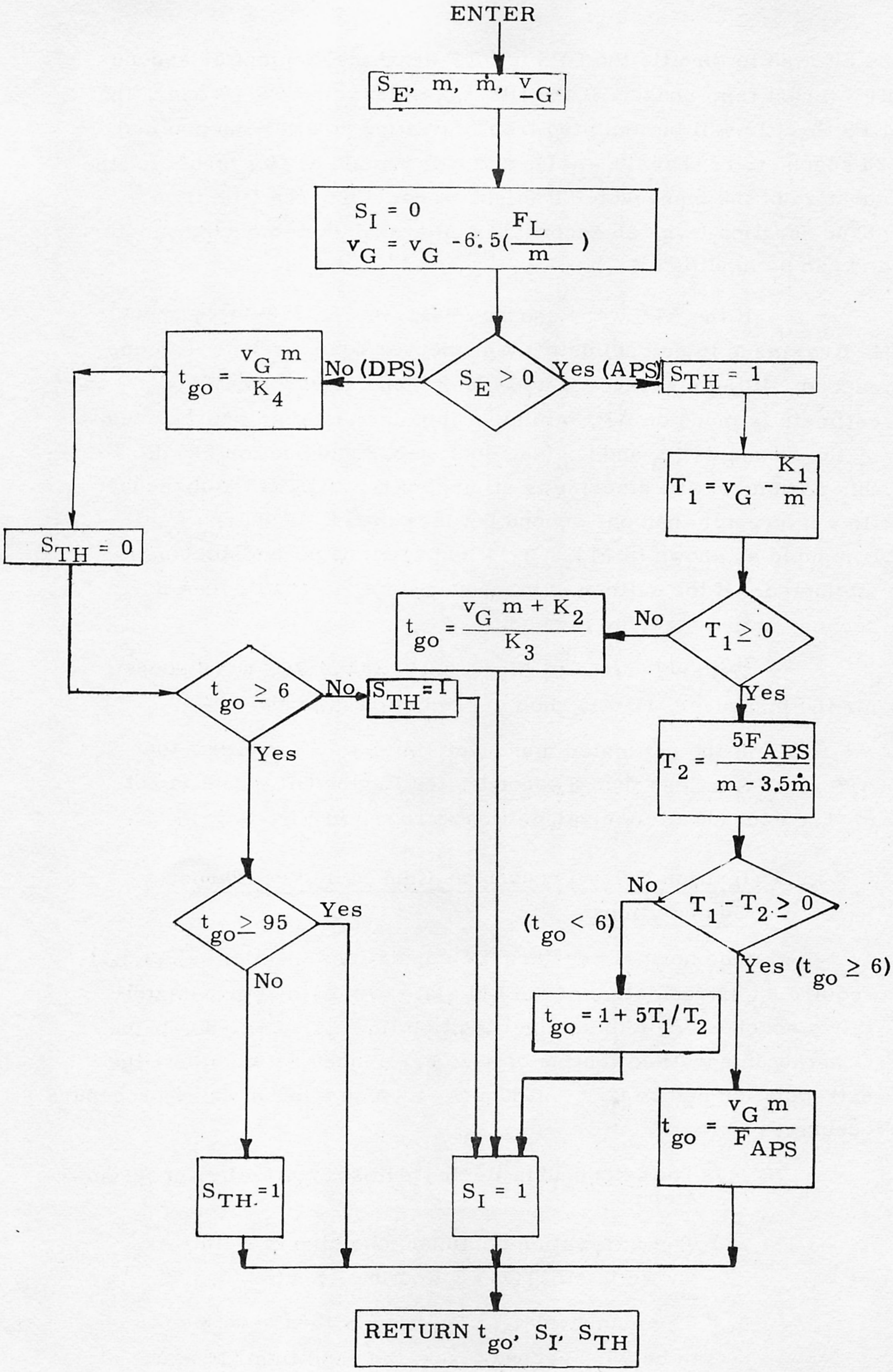


Figure 3.3-11 Time-to-go Predictor Routine

be allowed to throttle the DPS to FTP after the nominal 26 second 10% thrust trim phase. If the DPS t_{go} is less than 95 seconds, the DPS throttle will be inhibited from throttling up after the nominal 26 second trim phase (S_{TH}=1), and will remain at 10% thrust for the duration of the maneuver. It might be noted that the DPS trim phase duration (e.g. 26 seconds) is an erasable memory parameter and can be modified.

If the APS were chosen (P-42) in Fig. 3. 3-11, a check is first made to determine if the maneuver time, is less than one second. If the maneuver time is less than one second, the t_{go} estimate is made on APS minimum impulse test data represented by the constants K_2 and K_3 (See Section 5. 8 and Section 6). In this case no active steering is attempted (S_I = 1). If the maneuver time is greater than one second but less than 6 seconds, t_{go} is computed as shown in Fig. 3. 3-11 and again no active steering is attempted. If the estimated maneuver time is greater than 6 seconds active steering is used.

With reference to Fig. 3. 3-10, if the RCS were chosen for the maneuver, (P-41), no t_{go} prediction is made.

If the estimated maneuver time, t_{go}, for either the APS or DPS is less than 6 seconds, the Engine-Off signal is set for t_{go} seconds after actual ignition as shown in Fig. 3. 3-10.

5.3.3.3.4 Ignition Delay Procedures Caused by Pre-Thrust Computations

The normal pre-thrust computations (Section 5.3.3.3.1) require an extrapolation of the LM state vector to approximately thirty seconds prior to nominal ignition time, i. e. t_{IG}-30. If the Coasting Integration Routine of Sec. 5. 2. 2 does not complete the extrapolation before t_{IG} - 50 occurs, then an ignition delay procedure occurs as follows:

1.) The astronaut is alerted to this condition by a program alarm.

2.) The integration continues one step at a time as shown in MIDTOAVE Routine - Section 5. 3. 8.

3.) The maneuver ignition time is then redefined to to be 29. 9 seconds from the resulting LM state vector time, and the normal pre-ignition sequence is started.

5.3.3.4 Cross Product Steering Routine

The cross product steering concept is the basic control concept for both External ΔV and Lambert Aim Point guidance modes. The cross product steering logic is shown in Fig. 3.3-12. The following parameter definitions not previously described apply to this figure.

$\Delta \underline{v}$ — PIPA measured velocity vector over the computation cycle Δt transformed to the Basic Reference Coordinate System.

s_W — A logic switch in the cross product steering routine which when set to 1 allows steering commands and t_{go} calculations to be made. In the External ΔV mode, when set to 0, the velocity-to-be-gained, $\underline{v}_G$, is updated by $\Delta \underline{v}$, Section 5.3.3.5 . s_W is set to 1 in the ΔV Monitor Subroutine of Section 5.3.3.6 after engine thrust has been detected, and set to 0 when the computed time-to-go first becomes less than four seconds.

$\Delta t_{Tail\text{-}off}$ — A negative constant representing the duration of a burn at 40 percent thrust for DPS, and 100% for APS, equivalent to the tail-off impulse after the engine off signal is issued. It is initialized to one of 3 values from fixed memory:

$\Delta t_{Tail\text{-}off}(APS)$ for APS Maneuvers (100% thrust)
$\Delta t_{Tail\text{-}off}(DPS)$ for DPS maneuvers at 40% thrust*
$\Delta t_{Tail\text{-}off}(P70)$ for P70 DPS abort (100% thrust)

V_e — Engine exhaust velocity = $g\ I_{SP}$

EXTDELVFLG — External ΔV flag

1 = External ΔV maneuver
0 = Lambert Aimpoint Maneuver

*It is not necessary that these burns (i.e., non-P-70) will be made at 40% thrust.

[REFSMMAT]	Transformation matrix from the BRC system to the IMU or Stable Member Coordinate system (Section 5.6.3.4).
$\underline{u}_{FDP}$	Desired thrust acceleration vector in Platform Coordinates.

With reference to Fig. 3.3-12, the logic switch s_W is set to zero by the sequencing routine. When s_W is zero the only function of the cross product steering routine is to update the velocity-to-be-gained vector $\underline{v}_G$ with $\Delta\underline{v}$ when in the **External** ΔV mode.

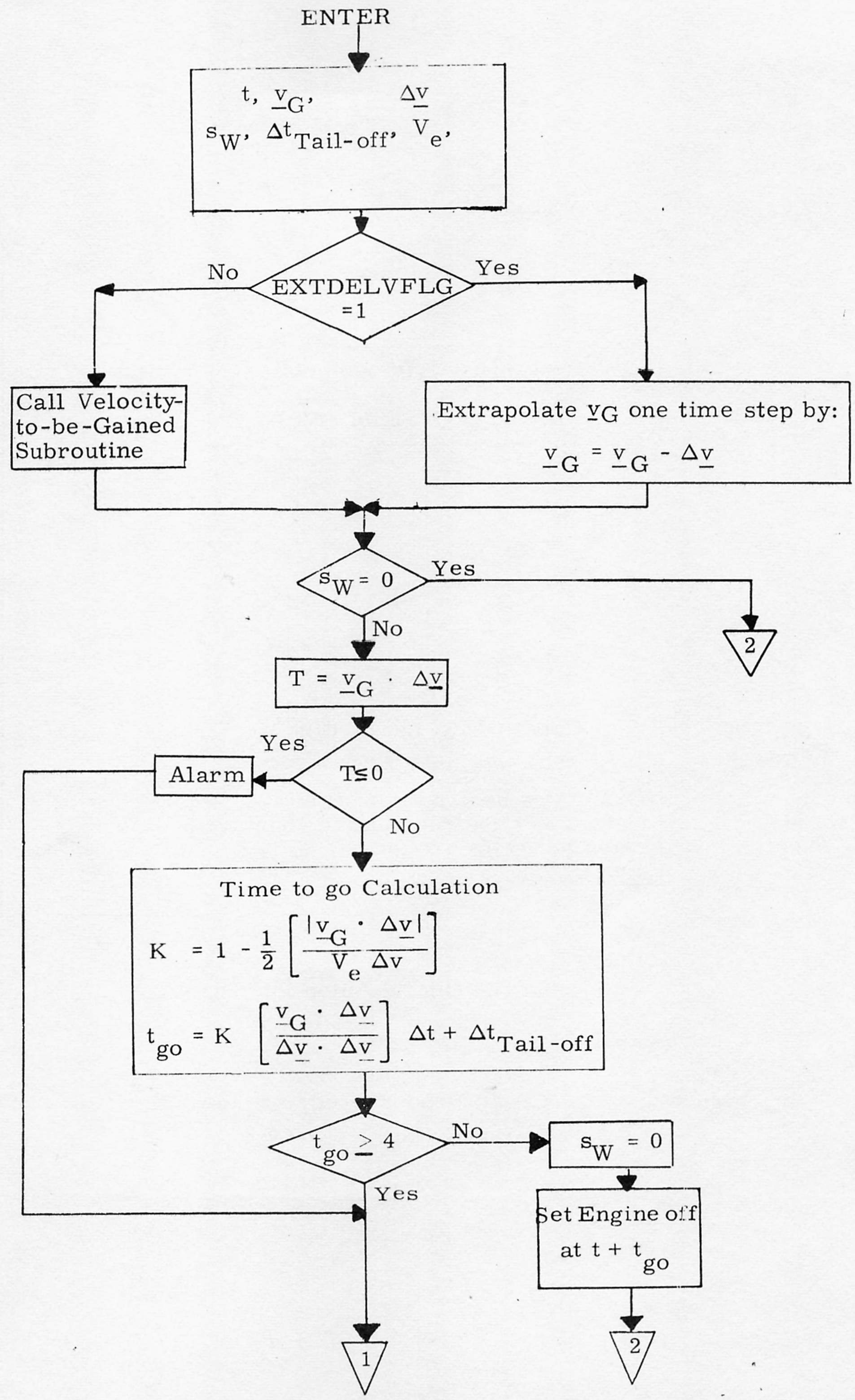


Figure 3. 3-12 Cross Product Steering Subroutine
(page 1 of 2)

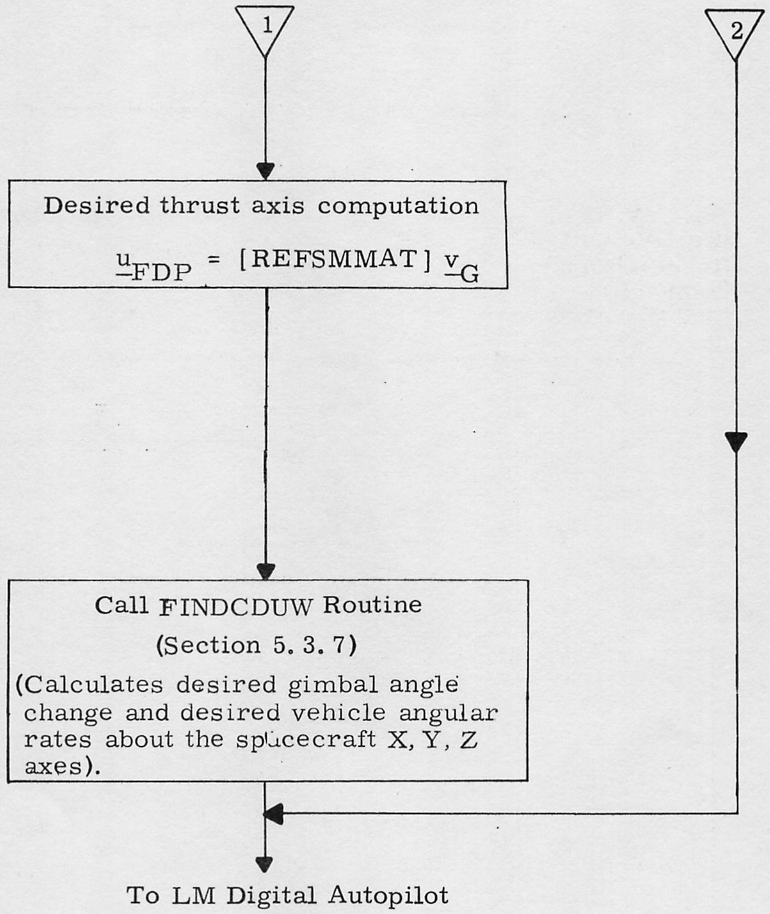


Figure 3.3-12 Cross Product Steering Subroutine
(page 2 of 2)

Switch s_W remains zero for short duration thrust periods or during the initiation of long duration thrust periods before the thrust has increased above the threshold level. In both these cases there is no active steering and the vehicle attitude is held at the pre-thrust alignment. When s_W is set to 1, active steering is performed. The time-to-go, t_{go}, computation and steering commands, $\underline{u}_{FDP}$, are performed as shown in Fig. 3. 3-12. When the computed t_{go} becomes less than 4 seconds, then the engine-off signal is set and switch s_W is reset to zero. For the remainder of the maneuver, no further computations are made except for v_G updating.

The steering command generated prior to calling the FINDCDUW Routine is the desired unit thrust direction vector in platform coordinates, $\underline{u}_{FDP}$. The FINDCDUW Routine of Sec. 5. 3. 7 is the interface routine between the guidance program and the LM Digital Autopilot (DAP) for all powered maneuvers. This interface routine generates three angle commands representing the desired change in IMU CDU gimbal angles (δ CDU), the desired angular rate vector, $\underline{\omega}$DV expressed in vehicle coordinates, and commanded attitude lag angles.

5.3.3.5 Velocity-to-be-Gained Routine

The velocity-to-be-gained computations shown in Fig. 3.3-13 are those carried out during the Lambert Aim Point powered flight guidance. The velocity-to-be-gained computation for the External ΔV guidance mode is simpler than that for the Lambert Aim Point guidance mode. The External ΔV velocity-to-be-gained computation is that shown in the cross product steering routine of Fig. 3.3-12 and is

$$\underline{v}_G = \underline{v}_G - \Delta\underline{v}$$

The velocity-to-be-gained computations for the Lambert Aim Point guidance mode involve the determination of a new $\underline{v}_G$ by processing the ASTEER Subroutine.

The following parameter definitions refer to Fig. 3.3-13.

$$\left.\begin{array}{l}\underline{r}(t)\\ \underline{v}(t)\\ t\end{array}\right\}$$ active vehicle state vector

$\underline{r}(t_2)$ Offset target intercept position vector at time t_2. This parameter is determined by the preceding targeting program (P-34, P35).

t_2	Intercept time of arrival associated with the offset target vector $\underline{r}(t_2)$. This is an input target parameter.
t_{IG}	Nominal ignition time.
t_0	Initialized to t_{IG}; used to test whether a new Lambert solution is needed, when $S_R = 0$.
$\underline{v}_R(t)$	Required velocity vector at time t.
$\underline{v}_G(t)$	Velocity-to-be-gained vector at time t.
S_R	Target rotation switch set in P-34, P-35 indicating that the target vector was rotated into the orbital plane due to proximity to 180° transfer.

$$S_R \begin{cases} 1 & \text{Target vector rotated} \\ 0 & \text{No rotation} \end{cases}$$

$\underline{g}_b$	Component of the earth gravity acceleration vector representing earth oblateness effects (Section 5.3.2).
U_T	Erasable time quantity used in determining whether a new "a" is needed.
s_G	$\begin{cases} 1 & \text{if transfer angle} < 180 \text{ deg.} \\ -1 & \text{if transfer angle} > 180 \text{ deg.} \end{cases}$
$\underline{h}$	Unit normal to the trajectory computed in the Initial Velocity Subroutine.

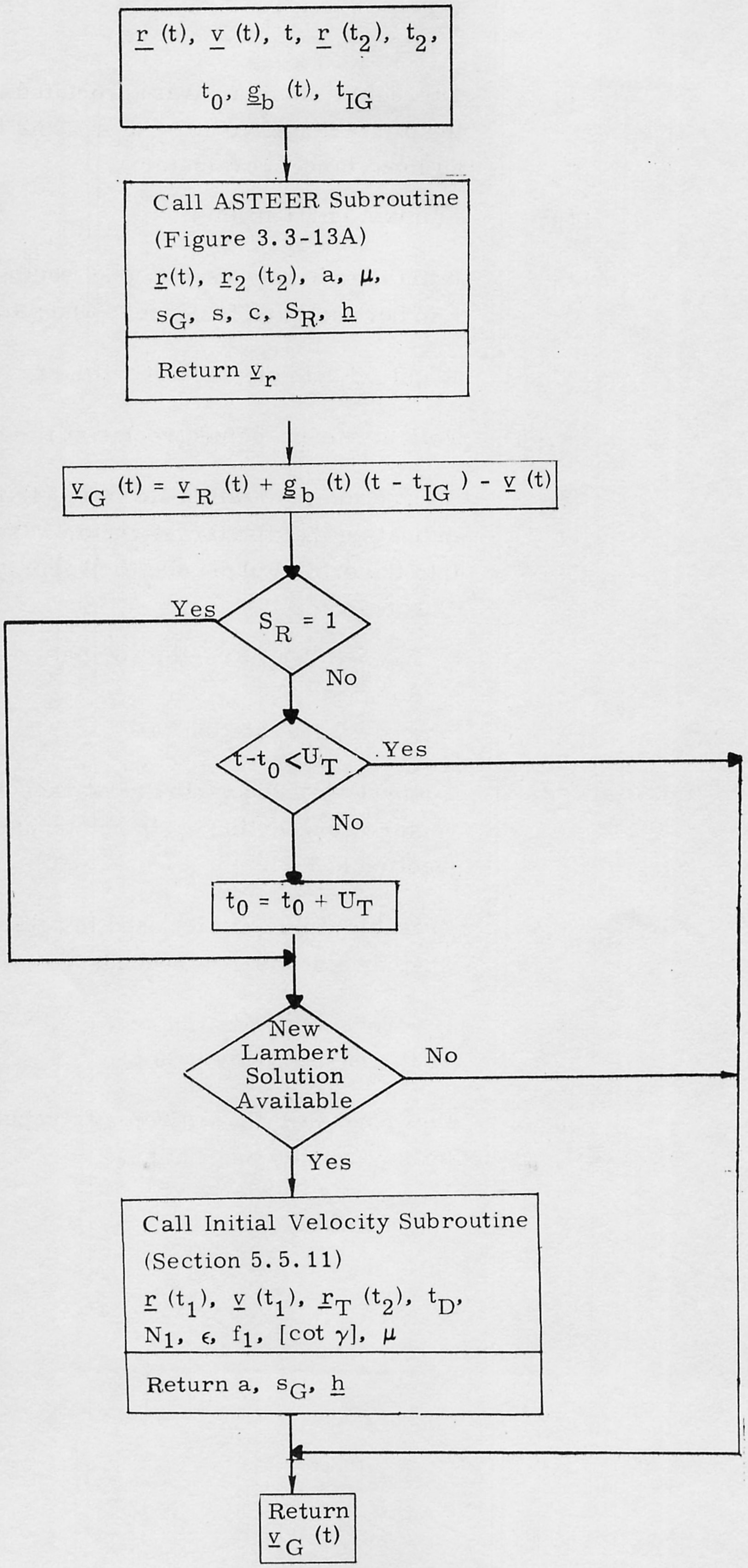


Figure 3.3-13. Velocity-to-be-Gained Subroutine

It should be noted that in Fig. 3.3-13 the velocity-to-be-gained, $\underline{v}_G$, derived from the ASTEER solution using an offset target vector is modified by the term $\underline{g}_b(t)$ $[t - t_{IG}]$. This term is an approximation to the velocity change contributed by the earth oblateness effect. The compensation used in this subroutine is computed as the current oblateness acceleration, $\underline{g}_b(t)$, multiplied by the time since nominal ignition $(t - t_{IG})$ where t_{IG} is the nominal ignition time. This correction is zero for lunar orbits. The objective of this correction is to reduce cut-off errors due to finite maneuver time effects, and to minimize commanded thrust attitude variations during the maneuver. These two effects occur during long maneuvers because in accounting for earth oblateness effects in the initial targeting programs (P-34 or P35), it is assumed that an impulsive maneuver will be applied at ignition time. Since a finite maneuver time is required, the precomputed target aim point becomes less accurate as the maneuver progresses. The $\underline{g}_b(t - t_{IG})$ correction is an approximate substitute for a retargeting procedure which can not be performed during a powered maneuver.

5.3.3.5.1 ASTEER Subroutine

The ASTEER Subroutine solves for the required velocity, ($\underline{v}_R$) given the semi-major axis parameter (a). Prior to ignition, the t_{IG} vehicle state vector is used in the Lambert Subroutine to obtain a semi-major axis (a) satisfying the time of flight and the conic aim vector. Then, starting at approximately t_{IG} -28 and at each guidance computation cycle (2 sec), the ASTEER Subroutine is called to determine an updated required velocity to satisfy the parameter a. Figure 3.3-13A presents the flow diagram for the $\underline{v}_R$ calculation. The following definitions refer to this figure.

$\underline{c}(t)$ chord vector $\underline{r}_2(t) - \underline{r}(t)$

$\underline{r}_2(t_2)$ conic aim vector

s semi-perimeter $\left[r(t) + r_2(t_2) + c(t)\right] / 2$

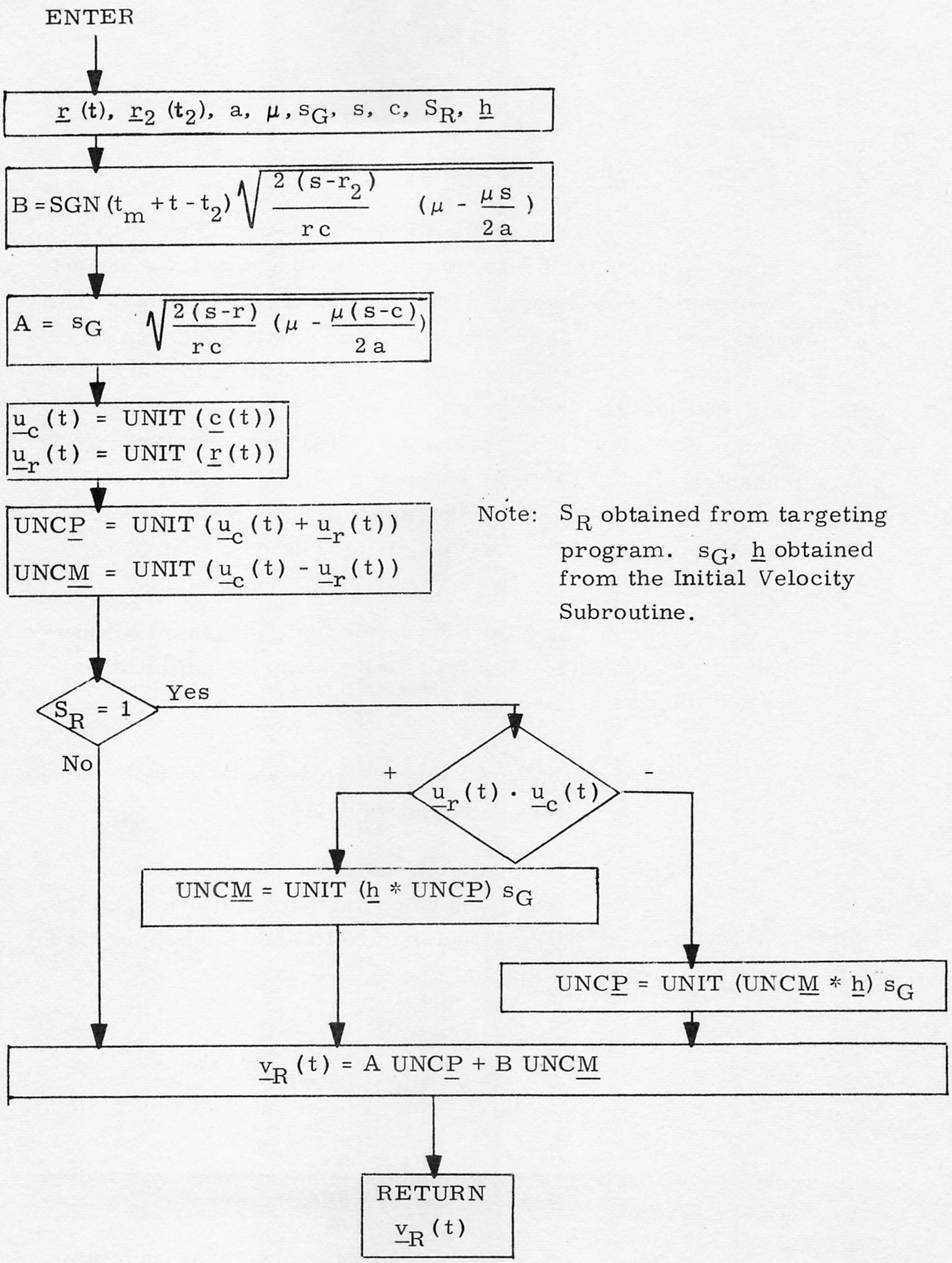
$$s_G \quad \begin{cases} 1 \text{ if transfer angle less than 180 degrees} \\ -1 \text{ if transfer angle greater than 180 degrees} \end{cases}$$

t_m minimum energy transfer time =

$$\sqrt{\left(\frac{s^3}{2\mu}\right)} \left\{\frac{\pi}{2} - s_G \left(\text{ARCSIN}\sqrt{\frac{s-c}{s}} - \frac{1}{s}\sqrt{c(s-c)}\right)\right\}$$

a semi-major axis obtained from Initial Velocity Subroutine

$\underline{h}$ Unit normal to the trajectory computed in the Initial Velocity Subroutine



Note: S_R obtained from targeting program. s_G, $\underline{h}$ obtained from the Initial Velocity Subroutine.

Figure 3.3-13A ASTEER Subroutine

5.3.3.6 ΔV Monitor Subroutine

The purpose of the ΔV Monitor Subroutine is to detect and monitor the presence of engine thrust every 2 seconds commencing at 2 to 4 seconds after the actual ignition time. The subroutine also turns on the cross product steering routine if the maneuver is not a short burn. If subsequent to 2 sec after the actual ignition time the engine thrust is detected to be below a given level for 5 cycles of this routine, control is transferred to the DPS/APS Thrust Fail Routine R-40. If during the thrusting period the engine thrust is detected to be below a given level for 2 cycles of this routine, control is transferred to the DPS/APS Thrust Fail Routine R-40.

The ΔV Monitor Subroutine logic diagram is illustrated in Fig. 3.3-14. The following parameter definitions apply to this section (5.3.3.6):

$\Delta \underline{v}$ — PIPA measured velocity change over the last computation cycle.

N_C — A counter in the ΔV Monitor Subroutine which indicates an engine failure when ≤ 0. This counter is initialized during pre-thrust computations to a value of 4.

S_I — Impulse Switch

$$S_I \begin{cases} 0 & \text{Allows steering at the proper time via } \Delta V \text{ Monitor Subroutine} \\ 1 & \text{Indicates short maneuver and no steering is required} \end{cases}$$

ΔV_K — A constant set by the pre-thrust routine which establishes the ΔV which must be sensed in a 2 second computation interval if the engine is to be considered on.

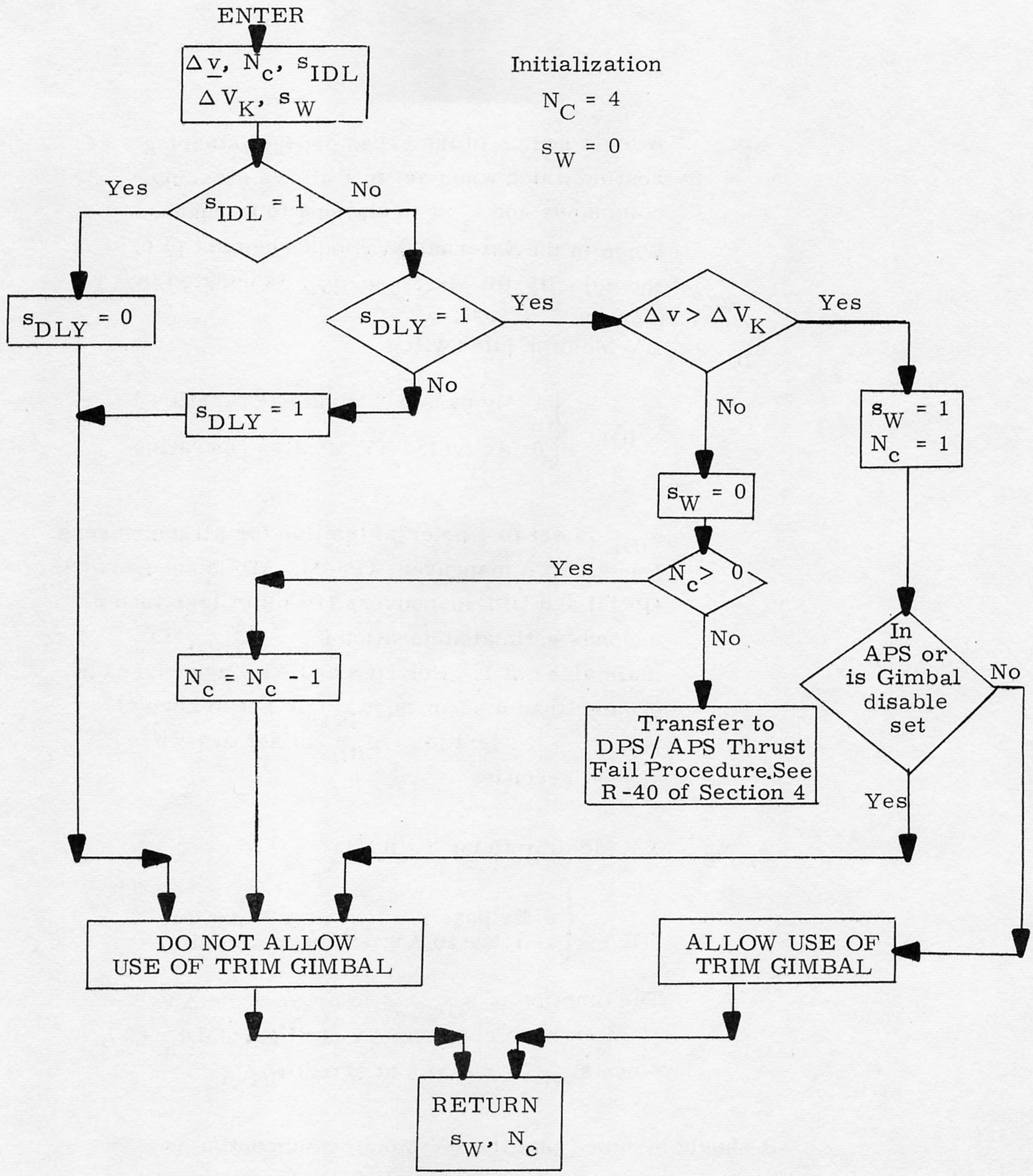


Figure 3.3-14 ΔV Monitor Subroutine

s_W A logic switch in the cross product steering routine which when set to 1 allows steering commands and t_{go} calculations to be made. When in the External ΔV mode when set to 0, the velocity-to-be-gained, $\underline{v}_G$, is updated by $\Delta \underline{v}$. (Section 5.3.3.4).

s_{IDL} ΔV Monitor Idle Switch

$$s_{IDL} \begin{cases} 1 & \text{By pass } \Delta V \text{ Monitor Operation} \\ 0 & \text{Activated } \Delta V \text{ Monitor Operation} \end{cases}$$

s_{IDL} is set to 1 prior to ignition for all maneuvers. For all RCS maneuvers (P-41), APS maneuvers (P-42) and DPS maneuvers (P-40) of less than 6 seconds estimated duration (S_I = 1) s_{IDL} is maintained at 1. For DPS and APS maneuvers of greater than 6 seconds, s_{IDL} is set to zero at ignition. s_{IDL} is set to 1 when $t_{go} < 4$ seconds.

s_{DLY} ΔV Monitor Delay Switch

$$s_{DLY} \begin{cases} 0 & \text{By pass } \Delta v \text{ test in } \Delta V \text{ Monitor} \\ 1 & \text{Activate } \Delta v \text{ test in } \Delta V \text{ Monitor} \end{cases}$$

The function of s_{DLY} is to prevent the ΔV Monitor from checking Δv until (actual $t_{IG} + 2$), since s_{IDL} is set to 0 at actual t_{IG}.

It should be noted that the ΔV Monitor Subroutine is activated by the Servicer Routine during all PGNCS controlled thrust maneuvers.* The engine fail procedures activated by this subroutine differ for various mission phase maneuvers as described in R-40 of Section 4.

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*The ΔV Monitor Subroutine is activated at (actual t_{IG} + 2) for DPS and APS maneuvers exceeding 6 seconds.

5.3.4 LUNAR LANDING GUIDANCE

5.3.4.1 Introduction

Lunar Landing Guidance is divided into two distinct parts:

1. Pre-ignition calculations (also called the Ignition Algorithm)
2. A guidance cycle starting before ullage initiation and ending after touchdown.

The pre-ignition calculations use the guidance equations as a subroutine. Therefore, the description and flow graphs of the pre-ignition calculations and the guidance cycle are grouped together.

The guidance cycle produces, in order (except for Auto P66):

1. A time of state vector validity (called PIPTIME)
2. The corresponding state vector*
3. Command thrust acceleration and window-pointing vectors, based on the state vector, according to the current phase
4. Throttle commands, based on command thrust acceleration
5. DAP commands, based on the command thrust acceleration and window-pointing vectors.

(For Auto P66 items 4 and 5 are interchanged)
Table 3.4.1-1 relates these computations to the routines where the computations are made.

Figure 3.4.1-1 illustrates the phases of the lunar landing. These phases are described as follows.

PRE-IGNITION

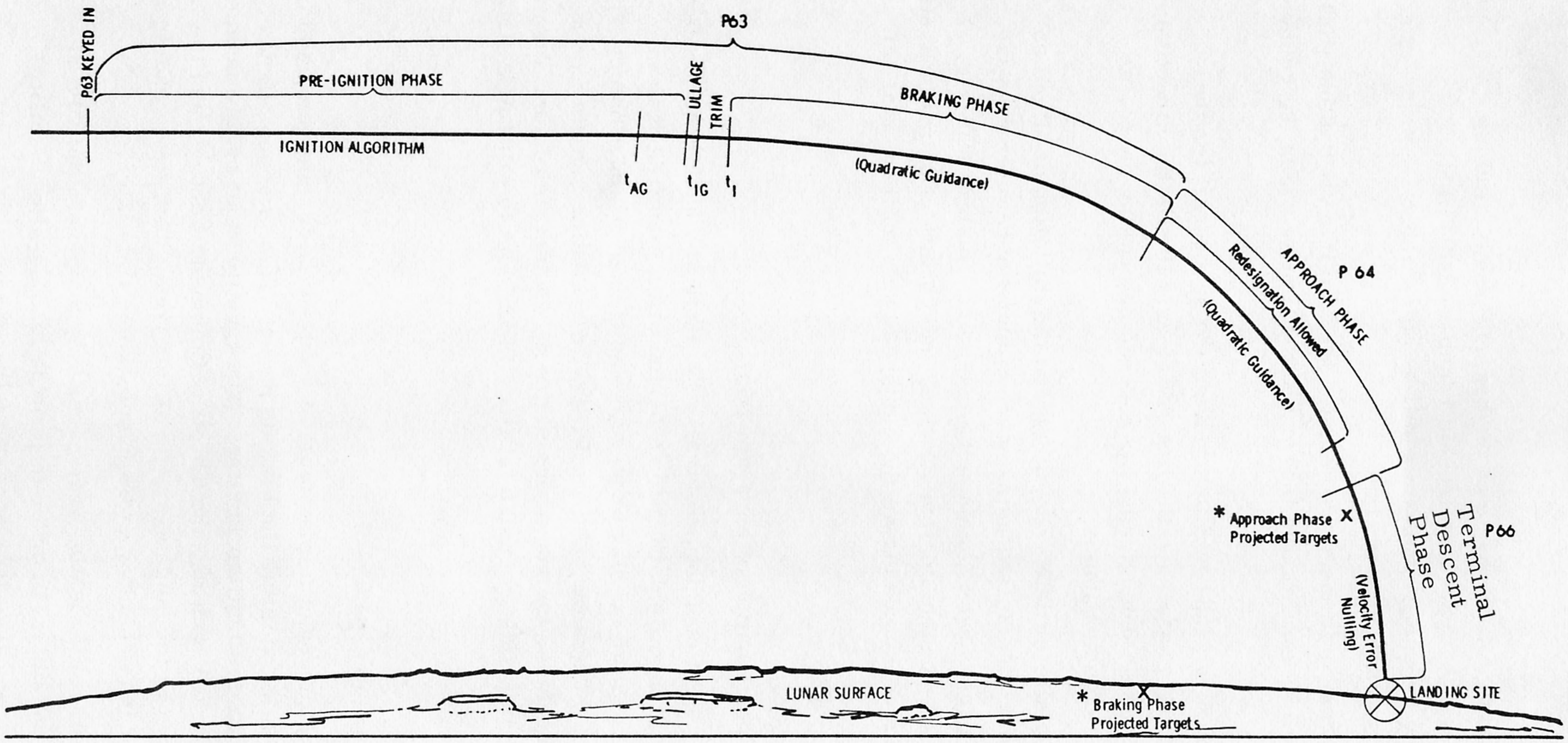
The astronaut starts the pre-ignition phase by selecting program P63 some time in the coasting descent orbit which precedes powered descent. The purpose of the pre-ignition phase is to determine ignition time based on the LM state in guidance coordinates. P63 determines the foregoing iteratively, while simultaneously determining the unit vectors defining the guidance frame, and the thrust acceleration and window-pointing vectors which

---

*The state vector is calculated using the following information: the previous state vector, readings from the PIPAs, estimated acceleration due to gravity, and readings from the radar (when radar readings are valid). See State Vector Update Routine.

Table 3.4.1-1 Guidance Cycle Computations and the Acting Routines

Calculation Of	Calculated By
State Vector	
a. from PIPA's	State Vector Update Routine
b. from Radar	LR Data Read Routine
Command Thrust Acceleration and Window-pointing Vectors	Ignition Algorithm and Guidance-and-Control Routine
Throttle commands	Throttle-Command Routine
DAP commands	FINDCDUW Routine



The above diagram, purposely drawn out of scale to fit on one page, illustrates the way the Lunar Landing Guidance Equations fit along the actual flight trajectory. The uncircled X's show approximate positions of projected targets for each phase. They are the targets that the spacecraft would achieve if the phase failed to end so that the same equations and targets were used until TTT reached zero.

*Targets achieved at these positions include position, velocity, acceleration, and one component of jerk. The X indicates the approximate position only; the braking phase targets may be located above or below the surface.

LUNAR LANDING PHASES

Figure 3.4.1-1

will be commanded on the first guidance cycle of the braking phase. During the pre-ignition phase, the vehicle travels in free-fall (unpowered) flight. Under normal conditions, the proper ignition time will be determined several minutes before the nominal ignition time.

After the above computations are complete, the MIDTOAVE Routine (R41) is entered to extrapolate the LM state vector forward to the time at which the guidance cycle is initiated.

The Attitude Maneuver Routine (R60) orients the spacecraft for the succeeding ULLAGE AND TRIM phase such that the LM X-axis lies in the direction of the first thrust command to be issued during the succeeding braking phase, as predicted by the pre-ignition computations.

The guidance cycle is initiated shortly before the initiation of ullage so that the LM state vector can be maintained using accelerometer measurements (average-G equations). The routines processed each guidance cycle are illustrated in Fig. 3.4.1-2, except that the Guidance-and-Control Routine, the Throttle-Command Routine, and the FINDCDUW Routine are excluded from the guidance cycle until the start of the braking phase.

ULLAGE AND TRIM

This phase is a continuation of program P63.

The LGC Master Ignition Routine starts RCS ullage and shortly thereafter ignites the engine at minimum throttle setting for a period of time sufficient to trim the thrust vector to point through the center of mass. The Guidance-and-Control Routine, the Throttle-Command Routine, and the FINDCDUW Routine remain excluded from the guidance cycle; therefore, the engine maintains a constant (minimum) throttle setting, and the spacecraft maintains a constant attitude throughout the ULLAGE AND TRIM phase.

BRAKING PHASE

This phase concludes program P63.

The throttle is commanded to maximum at the precise instant computed by the pre-ignition computations for starting the braking phase. This is accomplished by a WAITLIST call by the LGC Master Ignition Routine. The throttle must be advanced on time. The Guidance-and-Control Routine, the Throttle-Command Routine, and the FINDCDUW Routine are added to the guidance cycle at this time. The first reading (during the braking phase) of the accelerometers occurs some time later.

To utilize propellant efficiently, it is necessary to operate the DPS at maximum thrust as long as possible. In order to provide landing-site visibility during the succeeding approach phase, it is necessary that the braking phase

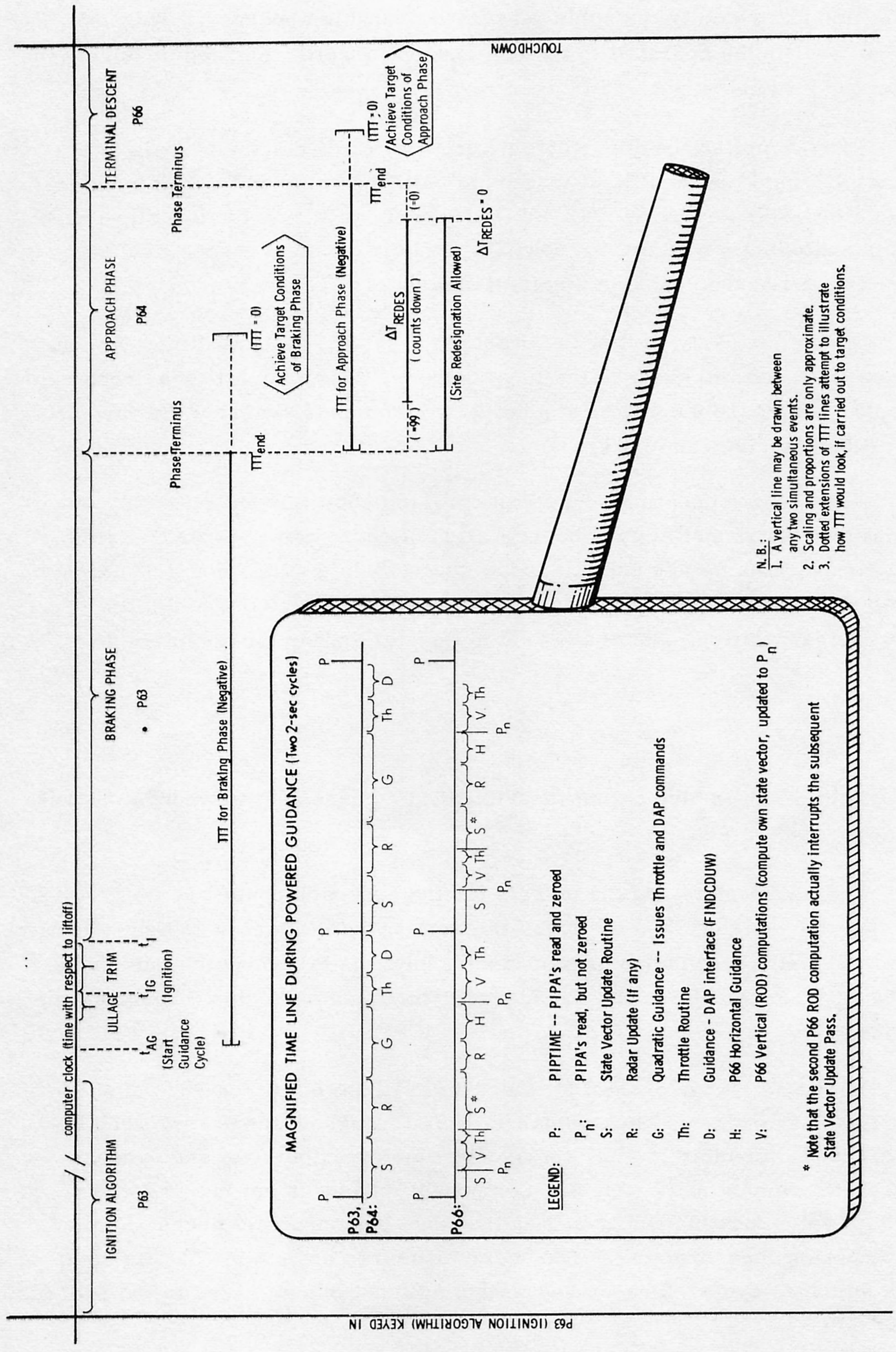


Figure 3.4.1-2. A Time Chart For Lunar Landing Guidance

targets (position and velocity) be achieved with reasonable accuracy. This requires that the DPS be operated in the continuously throttleable region during the final part of the braking phase.

To accomplish the above objectives, a throttle-command logic is employed which will hold the throttle at maximum until the command thrust acceleration is less than a pre-selected value (FLO). By properly selecting the braking-phase targets, this switching (or throttling-down) is made to occur at a specified time before the end of the phase on a nominal trajectory.

During the early part of the braking phase, the astronaut may yaw the vehicle about its X-axis to observe the lunar terrain. When the altitude drops below a pre-selected value, the guidance automatically commands a window-up yaw attitude to ensure proper landing radar operation.

State-vector updatings during the braking phase are based solely on IMU data until the astronaut allows incorporation of radar data. After this time, LR altitude measurements are used to update the vehicle's state each guidance cycle. Velocity measurements are also used when the estimated speed falls below a pre-selected value. See State Vector Update Routine for complete description.

APPROACH PHASE

This phase is also called the VISIBILITY phase and is synonymous with program P64.

The initial state and the targets for the approach phase are chosen so that the vehicle's X-axis (i.e., the thrust vector) is elevated sufficiently high above the local horizontal to permit landing-site visibility for the bulk of the phase. The DPS is operated in the continuously-throttleable region throughout the approach phase.

The state vector of the LM is updated with both altitude and velocity-component LR measurements each guidance cycle throughout the phase. The updatings take place immediately after the accelerometer output data are processed. An altitude measurement and a velocity-component measurement are processed sequentially at each updating time, with the altitude measurement preceding the velocity-component measurement. The radar measures any given velocity component every third guidance cycle. Pre-stored weighting functions are used in the processing of the radar data.

The astronaut has the option of manually redesignating the landing site by means of landing point designator (LPD) switch commands. These commands are computer inputs from the rotational hand controller. Each guidance cycle, a

computation is made of the new landing site location. The depression angle of the line-of-sight to the current site below the vehicle's Z-axis is computed and displayed to the astronaut throughout the approach phase. Normally, the current site is visible in the window, and the attitude commands are generated to yaw the vehicle about its X-axis until the line-of-sight to the current site is in the plane of the vehicle Z - and X-axes; i.e., it is along the LPD index line. Using the computed site depression angle on the DSKY, the astronaut can look out the window and decide where to redesignate the landing site, if he so desires.

TERMINAL DESCENT

P66 [also called Landing Phase Program] is entered automatically at the conclusion of P64. P66 can also be entered at any time after initial throttle-up by switching to Attitude Hold and manipulating the Rate of Descent Controller.

There are two parallel sections of P66, allowing for four possible combinations of PGNCS control:

VERTICAL	Auto Throttle Manual Throttle
HORIZONTAL	Auto Mode Attitude Hold Mode

When in Auto Throttle, P66 issues commands to the DPS such as to maintain a reference altitude rate. When P66 is entered, or whenever the throttle mode is manual, this reference rate is set to the current altitude rate. When in Auto Throttle, the reference rate can be manually incremented by manipulating the Rate of Descent Controller. When not in Auto Throttle, of course, the throttle commands are controlled by the astronaut.

When in Auto Attitude Mode--independently of the setting of the Throttle switch--P66 controls the attitude such as to null horizontal velocity. When in Attitude Hold, the astronaut controls attitude. If a PGNCS velocity error exists, the only way to null the true velocity relative to the moon is to switch to Attitude Hold. This is because there is currently no mechanism whereby the astronaut can correct the PGNCS velocity error.

5.3.4.2 Lunar Landing Coordinate Systems

Several coordinate systems are used for navigation and guidance of the LM during the powered landing maneuver. Each is a right-hand, orthogonal system. Figure 3.4.2-1 lists the various names by which each frame is known; it lists the mnemonic identifying initial used in this section; and it identifies the matrices used in this section and their equivalents from other sections, which transform between the coordinate systems.

The subscripts of these matrices are chosen according to the usual mathematical conventions which allow a product of transformations to be constructed by cancellation of internal subscripts. That is,

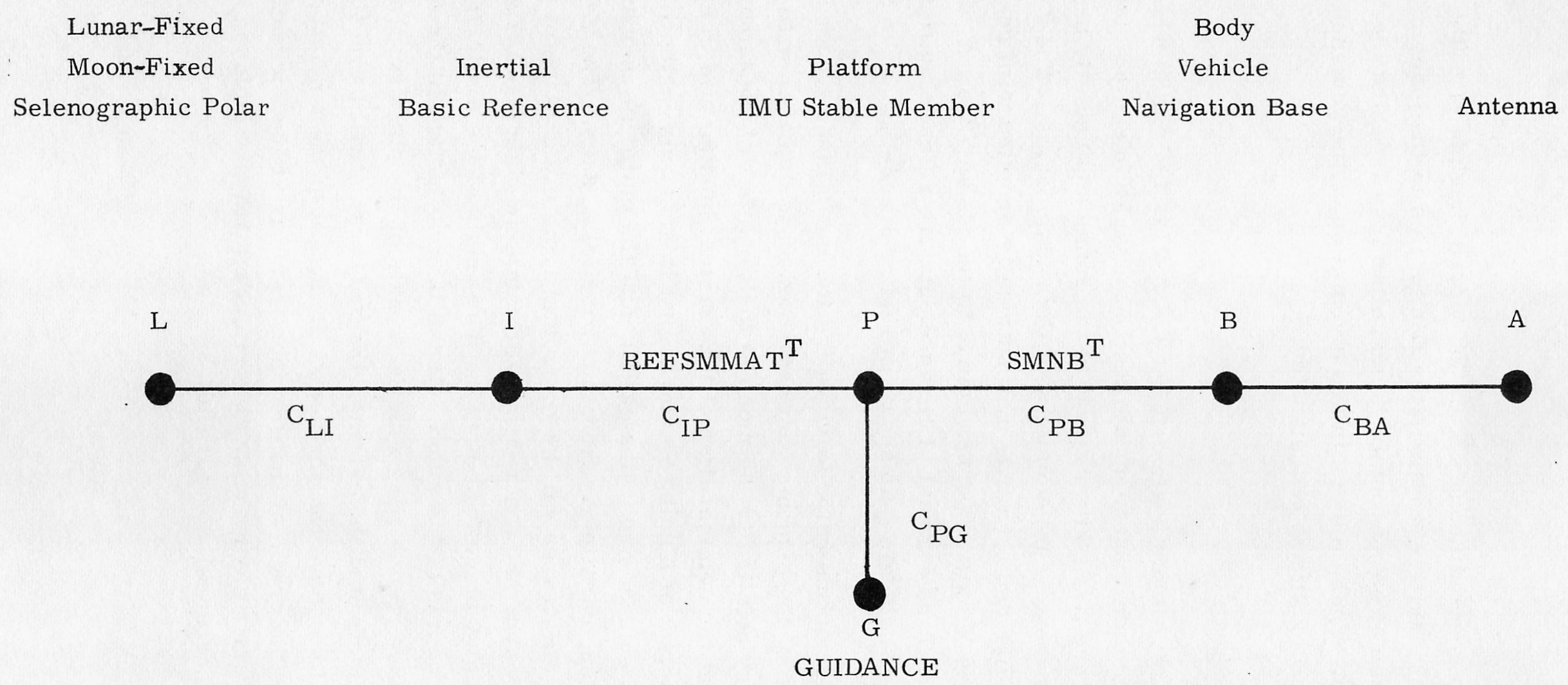
$$\underline{V}_L = C_{LA}\underline{V}_A = C_{LI}C_{IP}C_{PB}C_{BA}\underline{V}_A$$

where $\underline{V}_L$ is a column vector whose components are expressed in lunar-fixed coordinates, and $\underline{V}_A$ is the same vector in antenna coordinates. With this notation, the matrix C_{LA} is given by

$$C_{LA} = \begin{bmatrix} \underline{C}_{XLA} \\ \underline{C}_{YLA} \\ \underline{C}_{ZLA} \end{bmatrix}$$

where $\underline{C}_{XLA}$, $\underline{C}_{YLA}$, $\underline{C}_{ZLA}$ are the unit row vectors of the lunar-fixed coordinate frame expressed in antenna coordinates. These subscript conventions are opposite to those used in the Lunar Landing Guidance section of Revision 3 of the GSOP, Section 5.

Only those coordinate systems peculiar to landing navigation and guidance will be described in this section. The Lunar-Fixed, Inertial, Platform, and Body coordinates are described in the Coordinate Systems subsection of the Introduction to the GSOP; the transformations between Inertial, Platform, and Body coordinates are defined in the IMU Routines subsection of the General Service Routines, and the transformation between Lunar-Fixed and Inertial coordinates is defined in the Planetary Inertial Orientation Subroutine subsection of the Basic Subroutines.



Coordinate Frames, Aliases, and Transformations Between Adjacent Frames

Figure 3.4.2-1

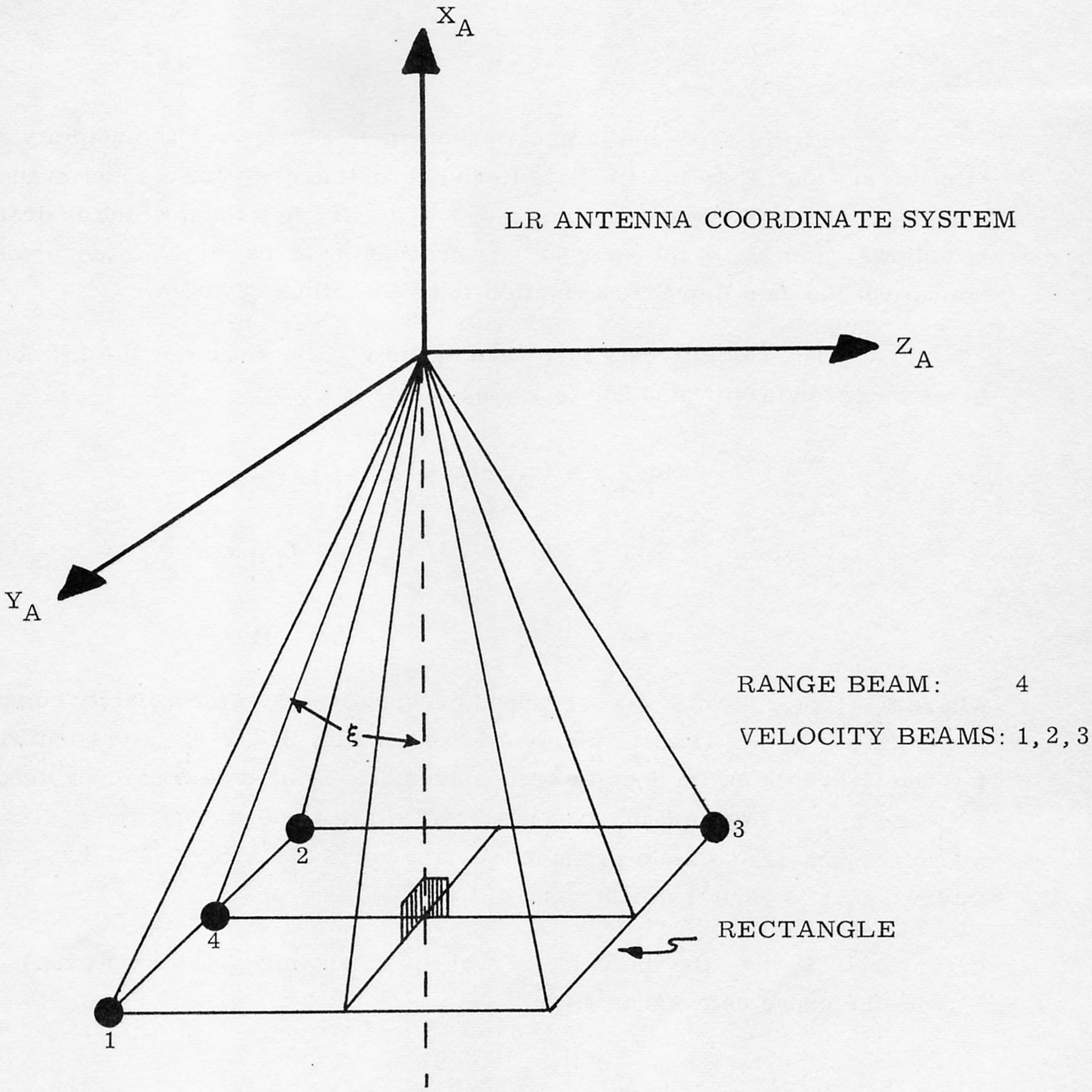
Guidance Coordinates

The origin coincides continuously with the current landing site (the frame rotates with the moon). The X-axis is vertical, the Z-axis lies in the plane of the trajectory relative to the moon at phase terminus and points forward, and the Y-axis completes a right-hand triad. Thus, the origin and orientation of the guidance frame are altered each time the landing site is redesignated.

Landing-Radar Antenna Coordinates

The LR antenna coordinate system (Fig. 3.4.2-2) is the one in which the range and velocity-component data used to update the LM navigation system are obtained. The antenna axes are fixed with respect to the range and velocity beams. The X-axis is oriented along the axis of symmetry of the radar beams. The Z-axis is normal to the X-axis and directed forward, symmetrically oriented with respect to the rear velocity beams. The Y-axis is perpendicular to the X- and Z-axes, and directed so as to form a right-hand system. LR range data are referenced to the LR antenna coordinate system by the angle ξ as shown in Figure 3.4.2-2.

The following description helps to visualize the lunar landing coordinate frames. Assuming the plane of the orbiting command module is coplanar with the plane of the LM trajectory relative to the moon at the terminus of the landing, and assuming the LM lands at the nominal time at the nominal landing site in a normal erect attitude, then at the instant of landing -- and at that instant only -- the LM body coordinates (B), the descent guidance coordinates (G), and the platform coordinates (P) will have collinear X-axes, parallel Y-axes, and parallel Z-axes. The X-axes will be vertical pointing up, the Z-axes will be horizontal lying in the plane of the two trajectories pointing forward, and the Y-axes will complete right-hand triads pointing normal to the trajectory plane to the right. The origin of the platform coordinates is always at the center of the moon. At the landing instant in the nominal case, the origin of the guidance frame will be on the X-axis of the platform frame at the intersection with the surface, and the origin of the body frame will be on the X-axis of the platform frame a few feet above the surface at the height of the landing radar antenna. At any other time, the origin of the guidance frame is at the currently computed landing site with the X-axis along the local vertical so that the guidance and platform X-axes are neither collinear nor parallel.



1) Range Beam 4 is in the X_A - Z_A plane at an angle ξ from the $-X_A$ axis

2) LR velocity data to LGC is in Antenna Coordinate System

3) LR Antenna Coordinate System is related to the PGNCS Navigation Base by a specified set of Euler angles for each of the two LR antenna positions (α, β)

Fig. 3.4.2-2 LR Antenna Coordinate System and Beam Configuration

LR Velocity and Range Measurement Data. -- The velocity data obtained from the LR by the LR Data Read Subroutine are with respect to the LR antenna coordinate system of Fig. 3.4.2-2 and are in a form which is described as follows, along with the various data processing steps the LGC performs to transform the data into the Navigation Base Coordinate System.

The velocity data furnished at the LGC interface by the LR comprise three binary data words of the following form:

$$S_{XA} = [(f_1 + f_3)/2 + f_B]\,\tau_{LR}$$

$$S_{YA} = [(f_1 - f_2) + f_B]\,\tau_{LR} \qquad (3.4.1)$$

$$S_{ZA} = [(f_3 - f_2) + f_B]\,\tau_{LR}$$

where S_{XA}, S_{YA}, and S_{ZA} correspond, respectively, to the velocity components along the $-X_A$, $+Y_A$, and $+Z_A$ antenna axes of Fig. 3.4.2-2. The quantities f_1, f_2, and f_3 are the beam doppler frequencies, f_B is the bias frequency used in the LR, and τ_{LR} is the time interval used by the LR when counting the cycles of the above frequencies so as to produce the data words S_{XA}, S_{YA}, and S_{ZA}. The time interval τ_{LR} is 80.001 milliseconds.

In the LGC the velocity along each antenna coordinate axis is computed from the above data words as follows:

$$v_{XA} = k_{XA}(S_{XA} - f_B\,\tau_{LR})$$

$$v_{YA} = k_{YA}(S_{YA} - f_B\,\tau_{LR}) \qquad (3.4.2)$$

$$v_{ZA} = k_{ZA}(S_{ZA} - f_B\,\tau_{LR})$$

where v_{XA}, v_{YA}, and v_{ZA} are the LR measured velocities along the positive antenna coordinate axes, and k_{XA}, k_{YA}, and k_{ZA} are the corresponding scale factors used to obtain the above velocities in feet per second.

The LR velocity information expressed by v_{XA}, v_{YA}, and v_{ZA} must first be transformed from the antenna coordinate system (subscript A) to the Navigation-Base coordinate frame (subscript B). This requires that the antenna-axes unit vectors be determined in the Navigation-Base coordinate frame.

The required antenna-axes unit vectors ($\underline{u}_{XAB}$, $\underline{u}_{YAB}$, $\underline{u}_{ZAB}$) are stored in fixed memory. Separate values are stored for antenna position 1 and position 2. These vectors are functions of the Euler angles, α and β, used to define the orientation of the LR antenna axes with respect to the navigation base, as shown in Figure 3.4.2-3. The following equations define the velocity component antenna axes:

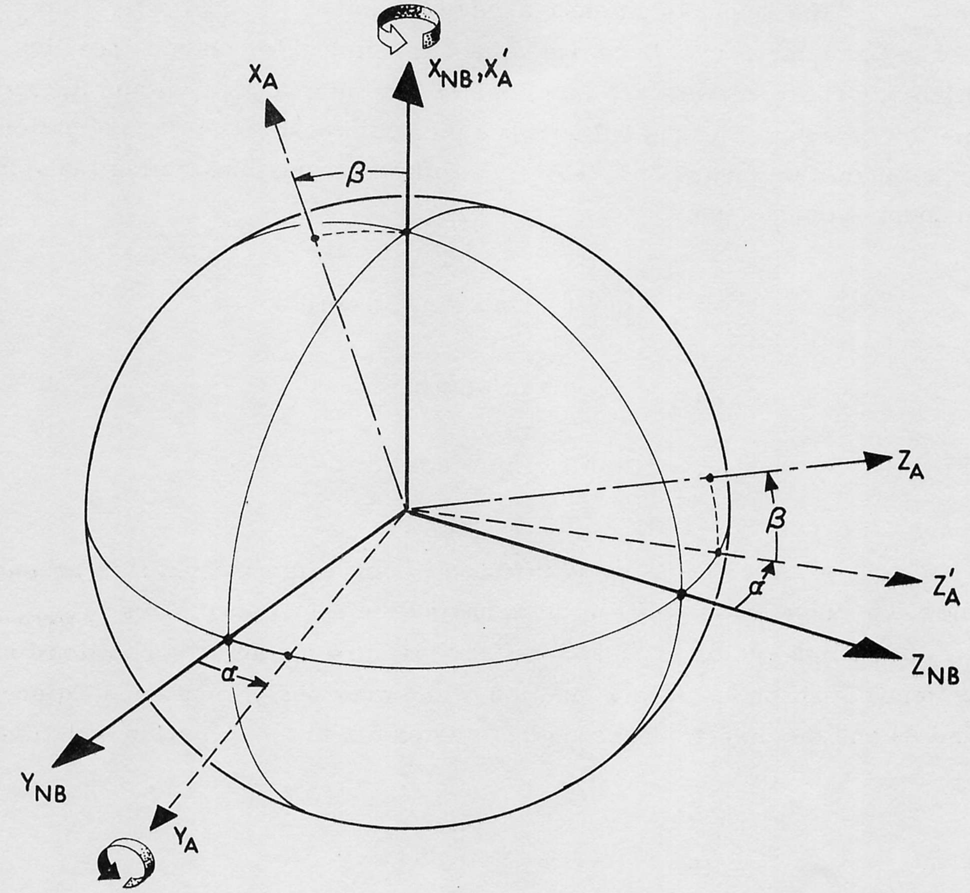
$$
\begin{aligned}
\underline{u}^T_{XAB} &= (\cos\beta,\ \sin\alpha\ \sin\beta,\ -\sin\beta\ \cos\alpha) \\
\underline{u}^T_{YAB} &= (0,\ \cos\alpha,\ \sin\alpha) \\
\underline{u}^T_{ZAB} &= (\sin\beta,\ -\sin\alpha\ \cos\beta,\ \cos\beta\ \cos\alpha)
\end{aligned}
\tag{3.4.3}
$$

In order to properly utilize LR range information it is necessary to know the range beam unit vector in navigation base coordinates ($\underline{u}_{RBB}$). The value of this vector is stored in fixed memory for antenna position 1 and for antenna position 2. It is a function of the previously mentioned Euler angles (α and β) and the displacement angle ξ; as shown in the following equations:

$$
\begin{aligned}
\underline{u}^T_{RBA} &= (-\cos\xi,\ 0,\ -\sin\xi) \\
\underline{u}_{RBB} &= \left[\underline{u}_{XAB} \,\middle|\, \underline{u}_{YAB} \,\middle|\, \underline{u}_{ZAB}\right]\underline{u}_{RBA}
\end{aligned}
\tag{3.4.4}
$$

The range beam is in the $X_A - Z_A$ plane of the antenna coordinate system, and is at an angle ξ from the $-X_A$ axis, as shown in Figure 3.4.2-2.

The range data obtained from the LR by the LR Data Read Routine is that measured by the LR along the range beam shown in Fig. 3.4.2-2. The range data is sent to the LGC from the LR as a binary data word R_{LR}, which represents the count of a certain frequency in the LR during the time interval τ_{LR}. Within the LGC, the range r_{LR} along the range beam is computed as follows:



ANGLE ORDER	AXIS OF ROTATION
α	X_{NB}
β	Y_A

Fig. 3.4.2-3 Angles Defining Orientation of LR Antenna Axes with Respect to the Navigation Base Coordinate System

$$r_{LR} = \begin{cases} k_{LR1}\ R_{LR} \\ k_{LR2}\ R_{LR} \end{cases} \qquad (3.4.5)$$

where k_{LR1} and k_{LR2} are the bit weights respectively for the long and short range scales in order to obtain r_{LR} in feet. When the LR Range Low Scale discrete is being received from the LR by the LGC, k_{LR2} is used.

A summary of the processing constants required by the LGC for LR operation is given as follows:

f_B	Velocity bias frequency.
τ_{LR}	Counting interval of the landing radar.
k_{XA}	Scale factor to convert ($S_{XA} - f_B\ \tau_{LR}$) to velocity along the LR antenna coordinate X_A (Fig. 3.4.2-2) in feet per second for the counting interval τ_{LR}.
k_{YA}	Scale factor to convert ($S_{YA} - f_B\ \tau_{LR}$) to velocity along the LR antenna coordinate Y_A in feet per second for the counting interval τ_{LR}.
k_{ZA}	Scale factor to convert ($S_{ZA} - f_B\ \tau_{LR}$) to velocity along the LR antenna coordinate Z_A in feet per second for the counting interval τ_{LR}.
$\underline{u}_{XAB}$, $\underline{u}_{YAB}$, $\underline{u}_{ZAB}$, $\underline{u}_{RBB}$	Unit vectors in direction of radar beam. (Separate values for Radar position 1 and position 2.)
k_{LR2}	Bit weight in feet for low range scale.

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5.3.4.3 State Vector Update Routine

The State Vector Update Routine computes the current state of the vehicle ($\underline{r}_P$, $\underline{v}_P$) using output data from the IMU and the landing radar (LR). Landing radar data (altitude and velocity) are included only if various tests are passed. The state-vector updatings occur at 2-second intervals during the landing maneuver, at the times that the PIPA outputs are processed. The state is first updated using IMU data, then LR altitude data. Finally, one of the three LR velocity components is processed. Thus the time between consecutive processings of the same LR velocity component is 6 seconds. The state-vector updatings are performed in the platform frame (subscript P).

The logic flow for the State Vector Update Routine is shown in Figure 3.4.3-1 through 3.4.3-6. First PIPA data are read and incorporated into the state vector. This incorporation is made into the temporary registers for position ($\underline{r}'_P$) and velocity ($\underline{v}'_P$). When the entire state vector update has been completed, these will be placed in the permanent registers ($\underline{r}_P$ and $\underline{v}_P$) that are available to the guidance and displays. Then h' is computed; this is the altitude of the spacecraft with respect to the landing site. h' is equal to the altitude above the center of the moon ($|\underline{r}'_P|$), minus the landing site radius (r_{LS}). The rotation of the lunar surface, $\underline{\text{VSURF}}$, as well as other parameters needed by guidance and displays is also computed at this time.

Preliminary LR computations are then entered, as shown in Figure 3.4.3-2. If the LRBYPASS flag is set (powered ascent and aborts), radar computations are bypassed. When the altitude first falls below 30,000 feet, the FLAUTOX and XORFLG flags are set (to inhibit x-axis override of the DAP). When the altitude falls below a cutoff value, HLROFF, the LRPERMIT flag is reset, to prevent further LR updates. Next, the NOLRREAD flag is checked. This flag is set during the time that the antenna is being repositioned. If set, the landing radar update is bypassed. The LR position test is now made. If neither or both discretes are present, the radar update is bypassed; and if this situation has existed for 10 seconds (5 passes) a 511 alarm is issued. (The 511 alarm will not be re-issued during subsequent passes if neither or both discretes remain present.) If the LR position discrete for position 1 or position 2 is present, it is stored. (Raw LR data will be processed by the fixed memory transformation matrix appropriate for the antenna position.) If the radar position has changed since the last pass, the radar update is bypassed, as the data may not be valid. If the position discrete has not changed, program flow proceeds to the LR Altitude Update.

Entered ~ 2 sec
at Priority = 20

Schedule LR Data Read Routine for Next Pass

Read PIPAs, Store Two Temporary Copies of PIPA Readings, and Record Time

$$\text{PIPTIME}' = t$$
$$\underline{\text{DELV}}_P = \underline{\text{PIPA}}_P$$
$$\underline{\text{TEM}}_P = -\underline{\text{PIPA}}_P$$
$$\underline{\text{PIPATMP}}_P = \underline{\text{PIPA}}_P$$
$$\underline{\text{PIPA}}_P = \underline{0}$$

Compute Guidance Period

$$\text{PGUIDE} = \text{PIPTIME}' - \text{PIPTIME}$$

Compensate PIPA Readings
(see IMU Compensation, Section 5.6.13)

Update Mass and Accumulated Velocity Increment

$$m' = m\,(1 - |\underline{\text{DELV}}_P|)\,/V_e(\text{DPS})$$
$$\Delta V_M = \Delta V_M + |\underline{\text{DELV}}_P|$$

Update State Estimate

$$\underline{r}'_P = \underline{r}_P + \text{PGUIDE}(\underline{v}_P + \underline{g}_P\,\Delta t/2 + \underline{\text{DELV}}_P/2)$$
$$\underline{g}'_P = -\mu_M\,\underline{r}'_P/|\underline{r}'_P|^3, \quad \underline{u}'_{hP} = \text{UNIT}\,(\underline{r}'_P)$$
$$\underline{v}'_P = \underline{v}_P + (\underline{g}_P + \underline{g}'_P)\,\Delta t/2 + \underline{\text{DELV}}_P$$
$$h' = |\underline{r}'_P| - r_{LS}$$
$\underline{\text{VSURF}} = \underline{\omega}_P \times \underline{r}'_P$ (rotation of moon's surface)
$\text{ABVEL} = |\underline{v}'_P|$ (for DSKY displays)

(1)

Figure 3.4.3-1. PIPA-Updates of State Vector

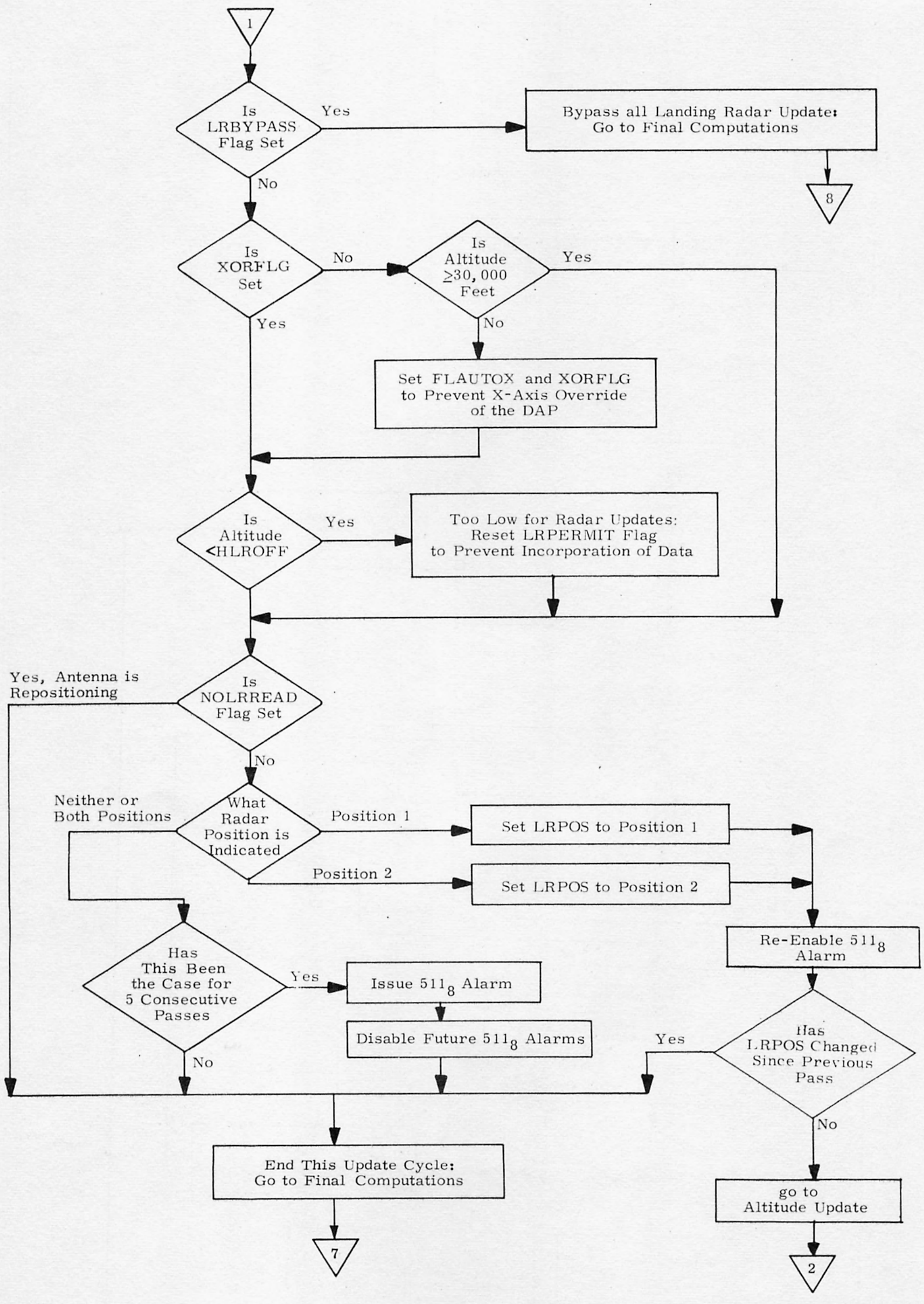


Figure 3.4.3-2 Preliminary LR Computations

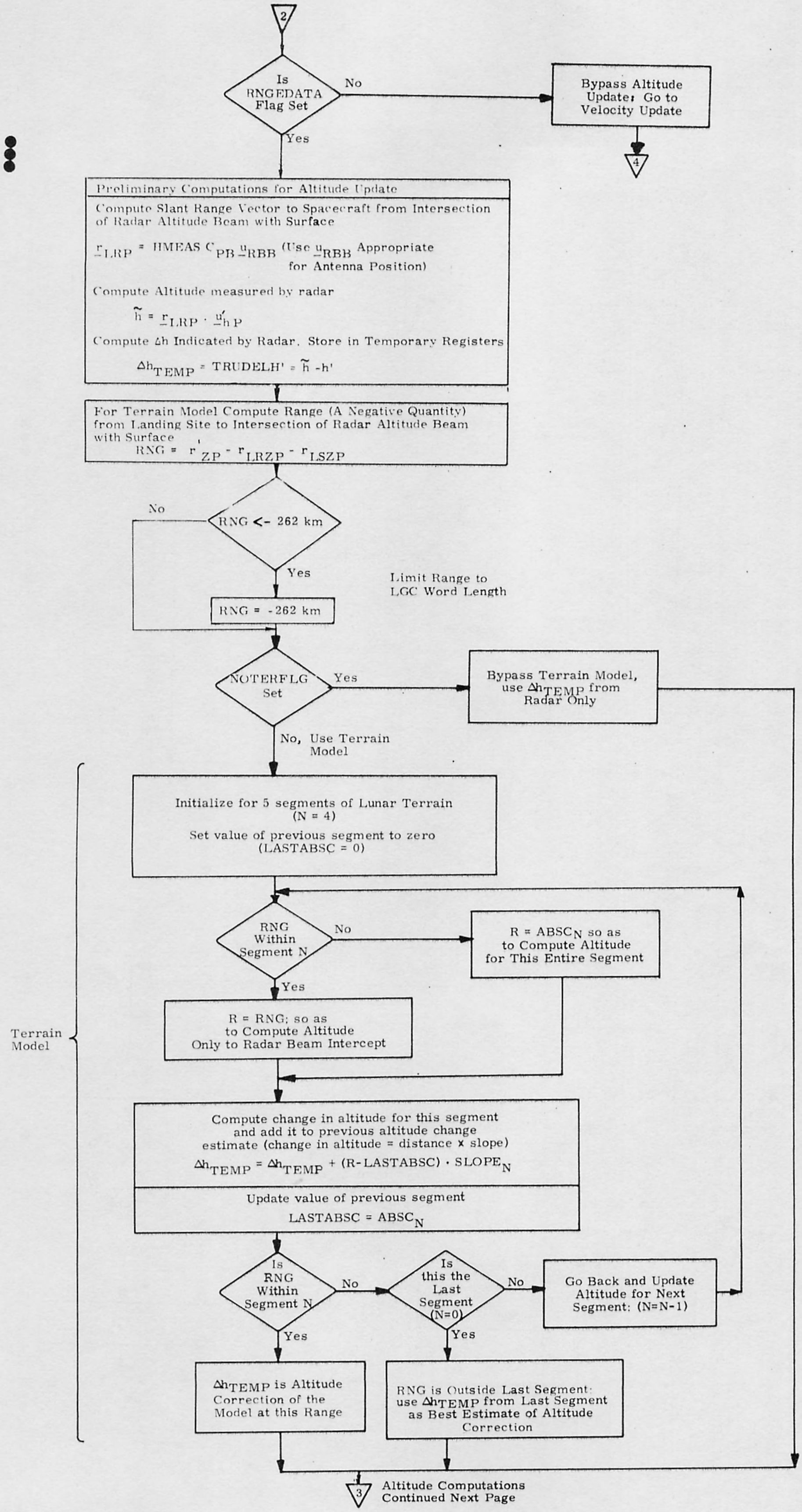


Figure 3.4.3-3 LR Altitude Update

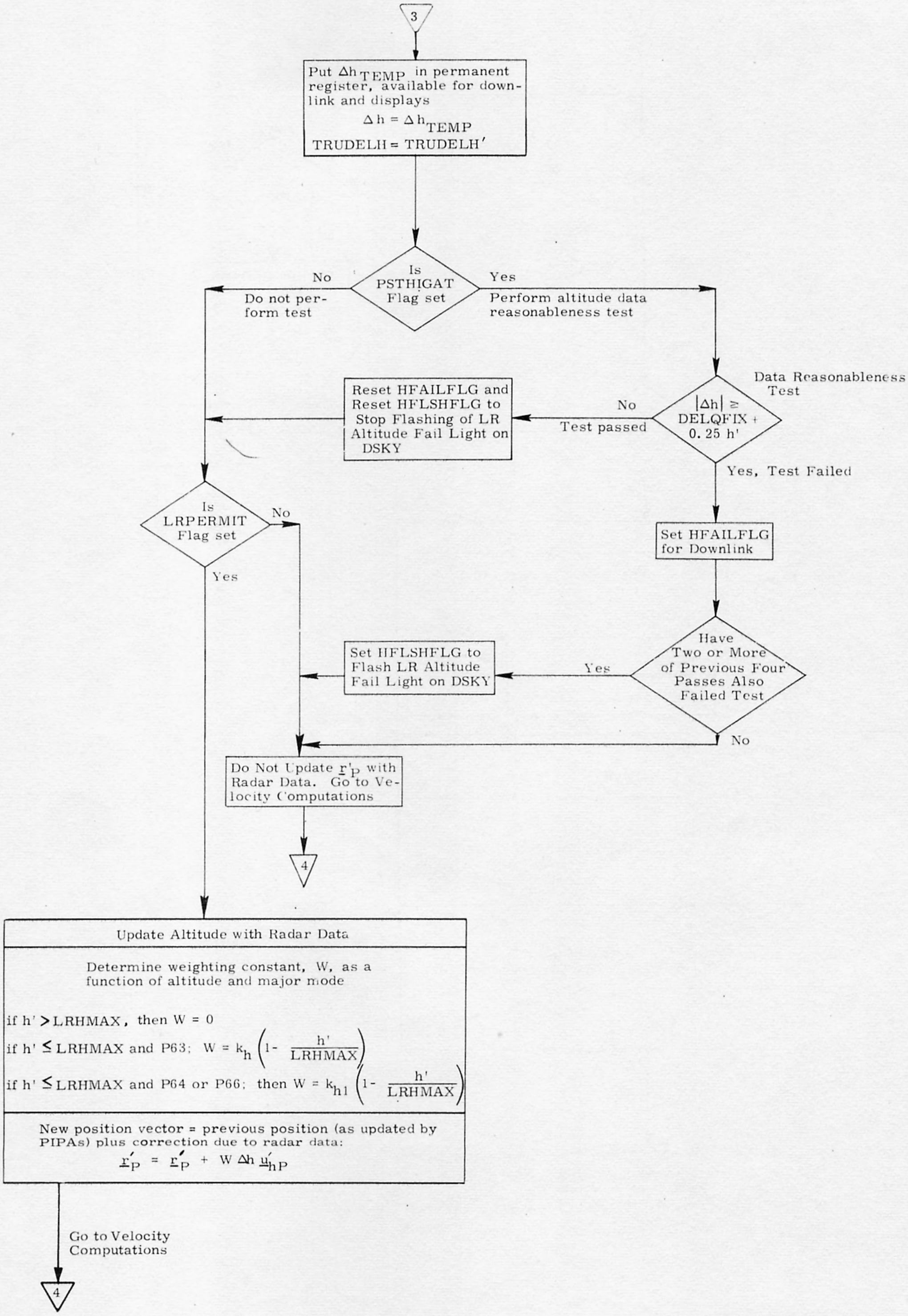


Figure 3.4.3-3 (Cont) LR Altitude Update

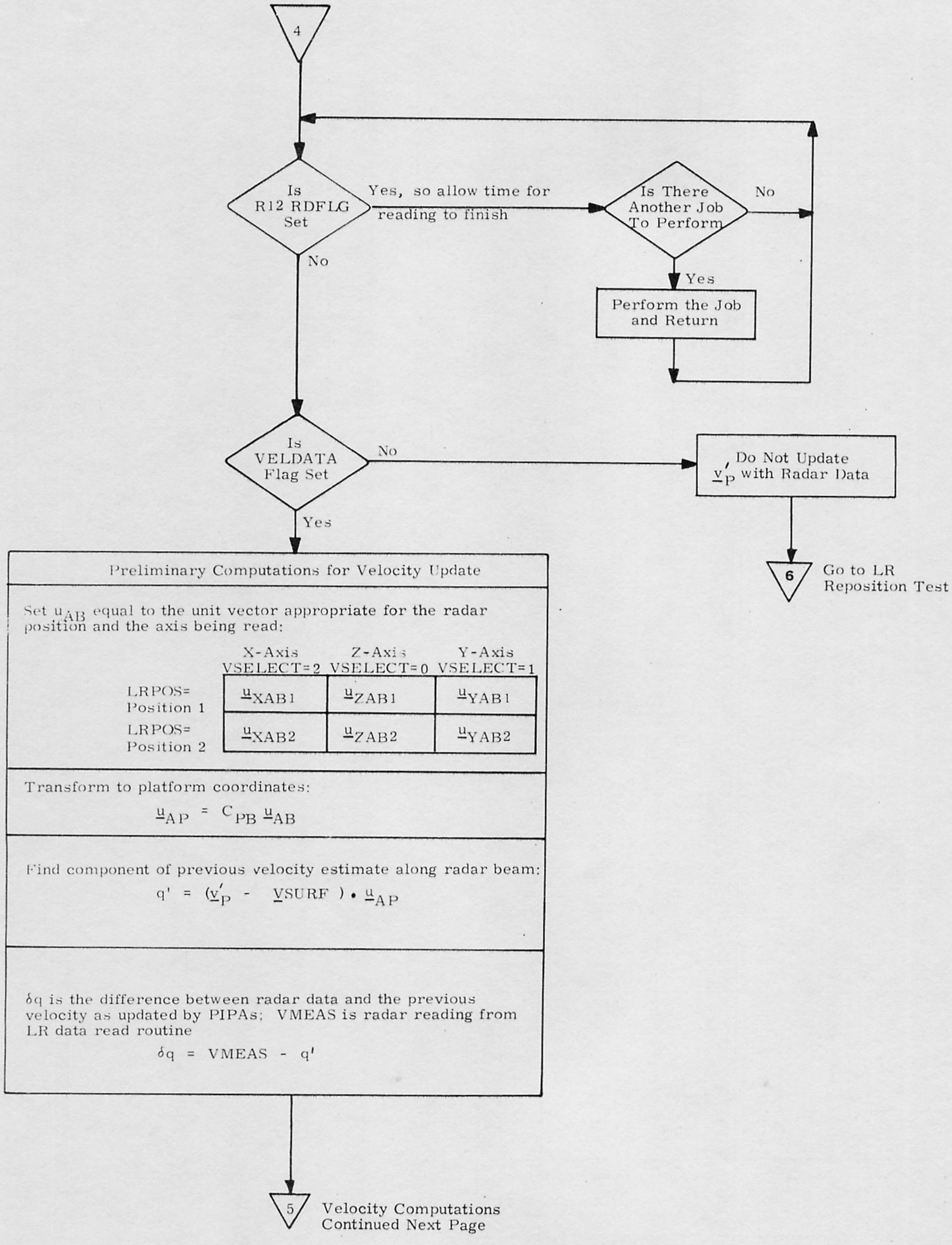


Figure 3.4.3-4 LR Velocity Update

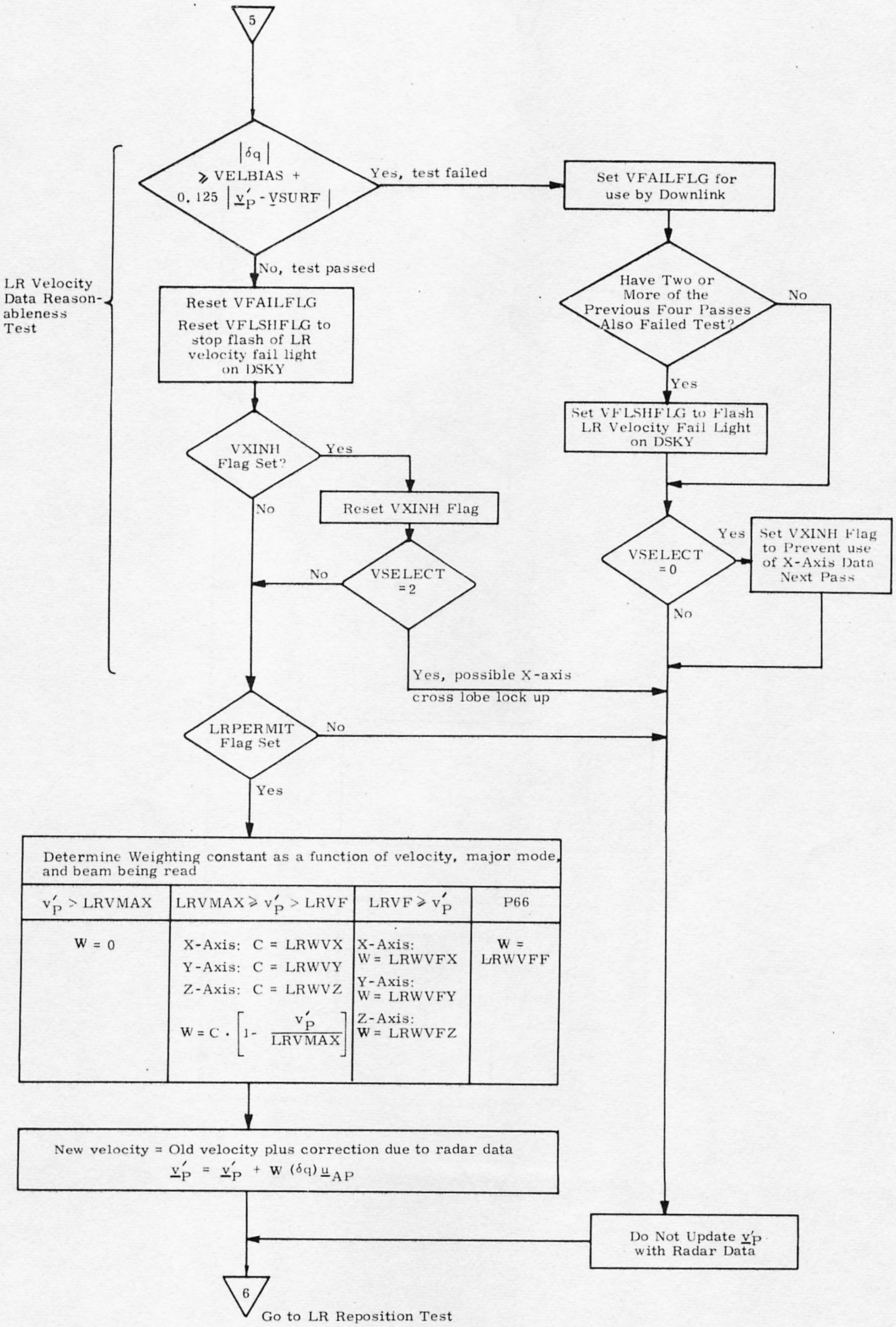


Figure 3.4.3-4 (Cont) LR Velocity Update

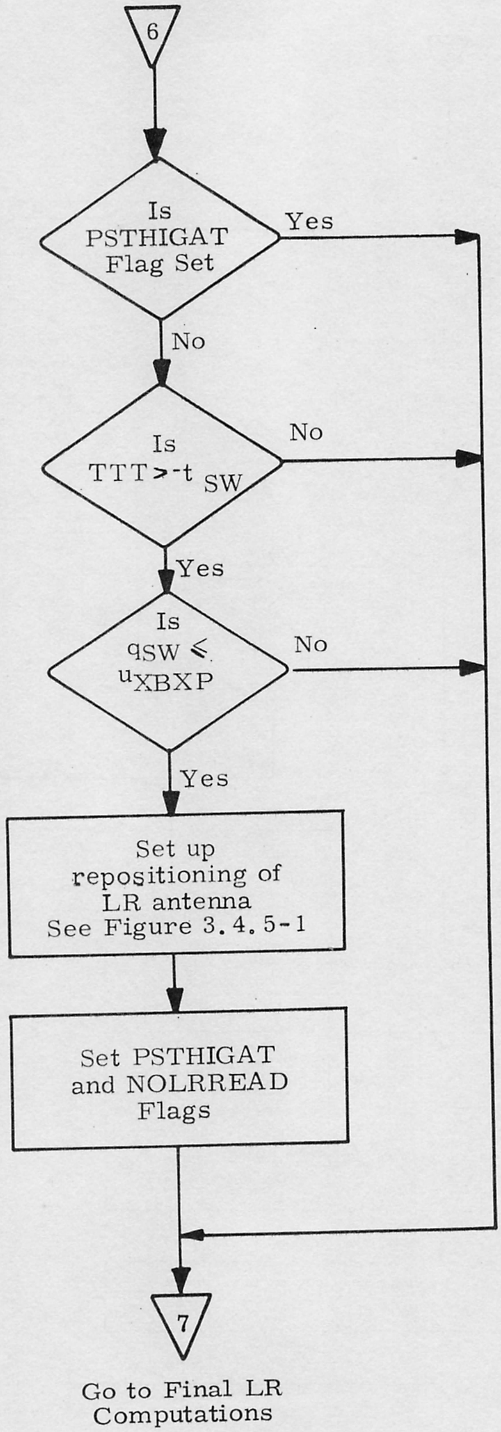


Figure 3.4.3-5 LR Reposition Test

7

Reset RNGEDATA and VELDATA flags

8

Update Altitude, Altitude Rate, and other parameters for Display

$$H' = \left|\underline{r}'_P\right| - \left|\underline{r}_{LS}\right| \qquad \dot{H}' = \underline{u}'_{hP} \cdot \underline{v}'_P$$

$$DALTRATE' = \frac{\left|\underline{u}'_{hP} \times \underline{v}'_P\right|^2}{\left|\underline{r}'_P\right|} \qquad (\underline{G} - \underline{V}BIAS)' = \underline{g}'_P - \underline{B}IAS$$

Update State Vector in Reference Coordinates

$$\underline{r}'_I = C_{IP}\ \underline{r}'_P \qquad \underline{v}'_I = C_{IP}\ \underline{v}'_P$$

Inhibit Interrupts to prevent use of partially updated data by other routines.

Set SWANDISP flag (Flag 7, Bit 11) to indicate that data is available for Landing Analog Displays

Transfer Data from the State Vector Update Routine to Permanent Registers while inhibit interrupt continues.

$\underline{r}_P = \underline{r}'_P$	$m = m'$
$\underline{v}_P = \underline{v}'_P$	$H = H'$
$\underline{r}_I = \underline{r}'_I$	$\dot{H} = \dot{H}'$
$\underline{v}_I = \underline{v}'_I$	$DALTRATE = DALTRATE'$
$PIPTIME = PIPTIME'$	$\underline{u}_{hP} = \underline{u}'_{hP}$
$\underline{g}_P = \underline{g}'_P$	$(\underline{G} - \underline{V}BIAS) = (\underline{G} - \underline{V}BIAS)'$

Zero PIPA Read Temporaries for P66 Velocity Update and Landing Analog Displays, while Inhibit Interrupt continues.

$$\underline{P}IPATMP_P = 0$$

Permit Interrupts, go to R13, Section 5.3.4.6

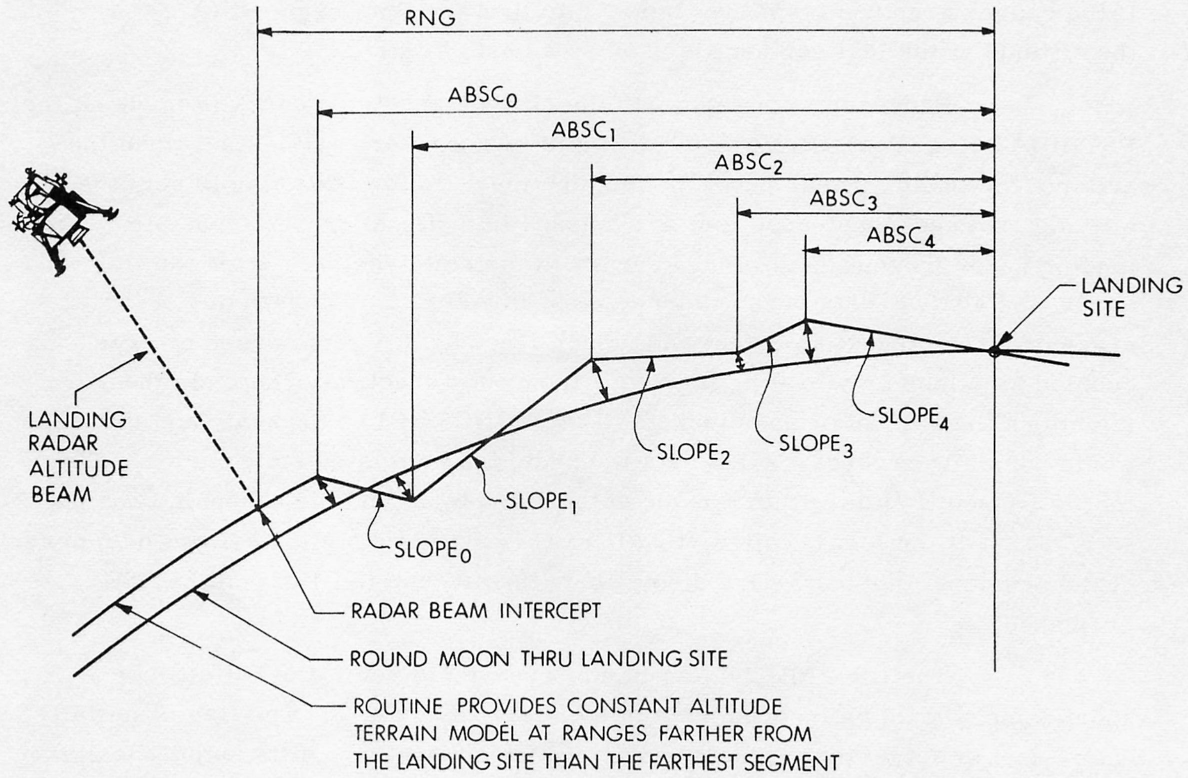
Figure 3.4.3-6. Final State Vector Computations

Altitude update (Fig. 3.4.3-3) will proceed only if the Range-measurement flag (RNGEDATA) is set. [This flag is set during each pass through the LR Data-Read Routine (Section 5.3.4.4), if the Range Data-Good discrete has been observed on this cycle and the previous two computation cycles. It is reset each pass, upon exit of the State Vector Update Routine.]

The actual measurement from the radar, HMEAS, is the slant range to the ground along the direction of the radar beam ($\underline{u}_{RBB}$). (Note that $\underline{u}_{RBB}$ depends upon antenna position.) This slant range is first transposed into platform coordinates using the vehicle-to-platform transformation matrix (C_{PB}). The resulting vector, $\underline{r}_{LRP}$, must then be projected along the local vertical so a valid comparison can be made with h', the PIPA-updated altitude. Thus $\tilde{h}$ is computed using $\underline{u}_{hP}$, the local vertical unit vector at PIPTIME.

Next computed is the discrepancy between radar (LR altitude) and PIPAs (the previous altitude as updated by PIPA data). This is stored as TRUDELH for downlink, and as Δh_{TEMP} for possible update by the terrain model. The change in altitude measured by the PIPAs reflects only inertial changes of the spacecraft. However, the radar measurement is affected by changes in the altitude of lunar terrain along the spacecraft trajectory, as well as inertial changes. During P63 and P64, the effects of terrain should be minimized, so that Δh can reflect the change in altitude above the landing site (change in h'). To accomplish this, the State Vector Update Routine uses a pre-stored model of lunar terrain along the nominal landing trajectory. During P66 the terrain model is turned off, since local variations in terrain are then important to the crew. A check is made on the NOTERFLG to see if the terrain model is to be used. If set (by the astronaut or by entrance to P66), the terrain model altitude incorporation is not made. (If the terrain model is not used, Δh_{TEMP} remains unchanged, as computed above.)

The terrain model uses a table of abscissas and slopes to define 5 line segments. These segments form a piecewise approximation of lunar terrain altitude along the landing trajectory. (See Fig. 3.4.3-7.) The routine



Note that range (RNG) and abscissas ($ABSC_N$) are stored as negative distances to landing site.

Fig. 3.4.3-7. Model of Lunar Terrain in LM Descent Trajectory

processes the altitude contribution of each segment, starting at the landing site. The contributions are added to the temporary register (Δh_{TEMP}) until the model is completed, so that Δh will be available for downlink and display. If the range (RNG) from the landing site to the radar beam lies outside a segment, the total altitude contribution of that segment is included, and the routine goes on to the next segment. If the range lies within a segment, the altitude to that point is computed, and the terrain model is concluded. Thus, as the vehicle approaches the landing site, segments are dropped off and the computations become shorter. If the range is farther from the landing site than the final segment ($ABSC_0$), the altitude at the last segment is used as a best estimate.

Proceeding forward with the altitude update, a check is made on the PSTHIGAT flag (Post-Higate). This flag is initially zero, and is set when the criteria for landing radar repositioning are met. Before this flag is set, the altitude data reasonableness test is bypassed in order to prevent possible landing radar lockout due to large errors in the state vector. After the flag is set, the altitude data reasonableness test is entered. The purpose of the reasonableness test is to detect and reject degraded LR data caused by cross-coupled side lobe or vibrating structure frequency-tracker lockup. In these situations the LR Data Good discrete to the PGNCS will be present, and the LGC would normally process the LR data to update the estimated state vector. The test is passed if Δh is within a value determined by h' and the erasable constant DELQFIX. If the test is failed, HFAILFLG is set for downlink. If two or more of the previous four tests have also failed, the altitude fail light on the DSKY is flashed.

The LR PERMIT flag is now checked to see if the LR altitude update, Δh, should be incorporated in the state vector, $\underline{r}'_P$. The flag is initially reset, and is set by the astronaut to permit incorporation. It is automatically reset when the altitude falls below the cutoff value, HLROFF. (See above.) If updating is permitted, a weighting function, W, is computed as a function of altitude; and the state vector is updated by the weighted Δh. If the flag is reset, radar updating is bypassed, and $\underline{r}'_P$ remains unchanged since the incorporation of PIPA data.

The LR velocity update is now entered (Figure 3.4.3-4). The R12 Read Flag (R12RDFLG) is checked to be sure all velocity readings have been

completed. If reading is in progress, velocity updating will wait for it to be completed. Velocity updating will take place only if the velocity-measurement flag (VELDATA) is set. [This flag is set during each pass through the LR Data-Read Routine (Section 5.3.4.4), but only if Velocity Data-Good discretes have been observed on this pass and on the previous two computation cycles. It is reset each pass, upon exit of the State Vector Update Routine.] If VELDATA is set, preliminary computations for velocity updating then take place. One of the three velocity components is computed each cycle. The axis to be computed (and thus the corresponding radar beam) is identified by the velocity component flag, VSELECT, from the LR Data-Read Routine. The unit vector in the direction of this radar beam ($\underline{u}_{AB}$) is selected. (The unit vectors for the different velocity components, referred to as $\underline{u}_{XAB}$, $\underline{u}_{YAB}$, and $\underline{u}_{ZAB}$, depend on the antenna position. See Section 5.3.4.2.) The selected beam vector is transposed into platform coordinates, using the body-to-platform matrix C_{PB}. For this pass, the vehicle velocity $\underline{v}_P'$ has been updated only by PIPA data. A component of this velocity is found along the direction of the radar beam (taking into account the rotation of the lunar surface, $\underline{V}$SURF). The difference between the present radar velocity and the PIPA-updated velocity, δq, is then computed; δq reflects the measured and the estimated velocity.

The velocity data reasonableness test is now entered to test for possible cross-lobe lockup. Note that the same erasable constant, VELBIAS, is used to test all three velocity beams. If the test is failed, velocity updating is prevented for this pass. Also, if the test is failed for a Z-axis read (VSELECT = 0), the VXINH flag (VX inhibit) will be set to prevent X-axis updating on the subsequent pass. This is because it is not always possible to detect cross-lobe lockup for the X-axis velocity component, although it can be detected for the Z-axis component.

As was the case for the altitude data reasonableness test described above, failure information is put on downlink, and if two of the four previous passes have also failed, the LR velocity fail light on the DSKY is flashed. It should also be noted that since the LR data reasonableness tests for altitude and velocity follow the LR Data-Good discrete checks in the LR Data-Read Routine, the data reasonableness alarm criterion is based on the last four LR readings which have passed the Data-Good discrete check. The LR data reasonableness tests are independent of intermittent tracking and do not account for any data rejected because of failure to pass the LR data good tests.

If the test is passed, the LRPERMIT flag is checked. (See description of this flag above.) If updating is permitted, a weighting function, W, is computed as a function of velocity, beam orientation, and major mode. The cutoff velocities LRVMAX and LRVF are in erasable memory as are the components of the weighting constant, LRWVX, LRWVY, LRWVZ, LRWVFX, LRWVFY, LRWVFZ, and LRWVFF. The state vector is then updated by the weighted value of δq along the direction of the velocity component $\underline{u}_{AP}$.

If the velocity data reasonableness test is failed or updating is not permitted (LRPERMIT = 0), radar updating is bypassed; and $\underline{v}'_P$ remains unchanged since the incorporation of PIPA data.

The LR reposition test is now entered (Fig. 3.4.3-5). If the PSTHIGAT flag is set, then the criteria for repositioning were met on a previous pass, and repositioning was attempted; therefore, the remainder of the test can be bypassed. Reposition criteria are the computed time-to-target, TTT, and the projection of the LM X-axis unit vector along the platform frame X-axis (u_{XBXP}). The test thresholds, t_{SW} and q_{SW} are stored in erasable memory. If both tests are satisfied, the PSTHIGAT flag is set. The NOLRREAD flag is also set to prevent radar updates for the duration of the repositioning. Repositioning is controlled by the LR position command subroutine (Section 5.3.4.5).

Final LR computations are now performed that prepare for the next pass through the State Vector Update Routine and store quantities needed by other parts of the lunar landing guidance cycle.

5.3.4.4 LR Data Read Routine

The LR Data Read Routine, Figure 3.4.4-1, reads and stores Landing Radar data necessary for state-vector updates. It is entered every ~2 sec via a scheduled call from the Descent State Vector Update Routine (R12). Upon entry, the routine initializes the various parameters used in the reading of radar data, initiates the first radar read, then exits and waits for the first radar interrupt.

The LR interrupt occurs 6 times during a 2-second pass, at 100 millisecond intervals. Each reading is the result of a 80.001 millisecond duration count of a frequency in the LR. One altitude, and five velocity readings are performed. The altitude reading is centered at PIPTIME (the time at which the PIPAs are read) and the velocity readings are distributed so that the average of the reading times is close to PIPTIME. This schedule of reading times permits a valid comparison between PIPA data and LR data in R12. When a radar interrupt occurs, the routine processes the appropriate data and prepares for the next interrupt, then exits. Thus, the radar processing is done in parallel with the state-vector update computations, guidance, and displays.

When the routine is entered, it first checks the NOLRREAD flag (set while the antenna is repositioning). If set, the routine is exited for this 2-second pass. If clear, initialization is performed. The R12RDFLG is set to prevent the processing of velocity data by the velocity update portion of R12 until the LR Data Read Routine is completed. The next step is to determine which velocity beam is to be read, according to the value of VSELECT. SAMPLSUM is set to zero; it will accumulate the 5 velocity readings during this 2-second pass. Bits 5 and 8 of channel 33, the altitude and velocity data-good discretes, are under the control of radar hardware; for these bits a 0 indicates acceptable radar data. The values of bits 5 and 8 are saved in OLDATAGD to reflect the status of the radar prior to the first reading. The altitude and velocity data fail flags (RADMODES, bits 5 and 8) are reset for use this pass. They will be set to 1 if the corresponding bit of channel 33 is 1 (i.e. radar data not acceptable). Next, the value of NSAMP is set to 6, and a velocity data-read is initiated to cause the first radar interrupt. Now the routine exits, and waits for the first radar interrupt.

NSAMP controls the order of altitude and velocity readings for the routine. For each interrupt, the reading is performed; then NSAMP is decremented and tested.

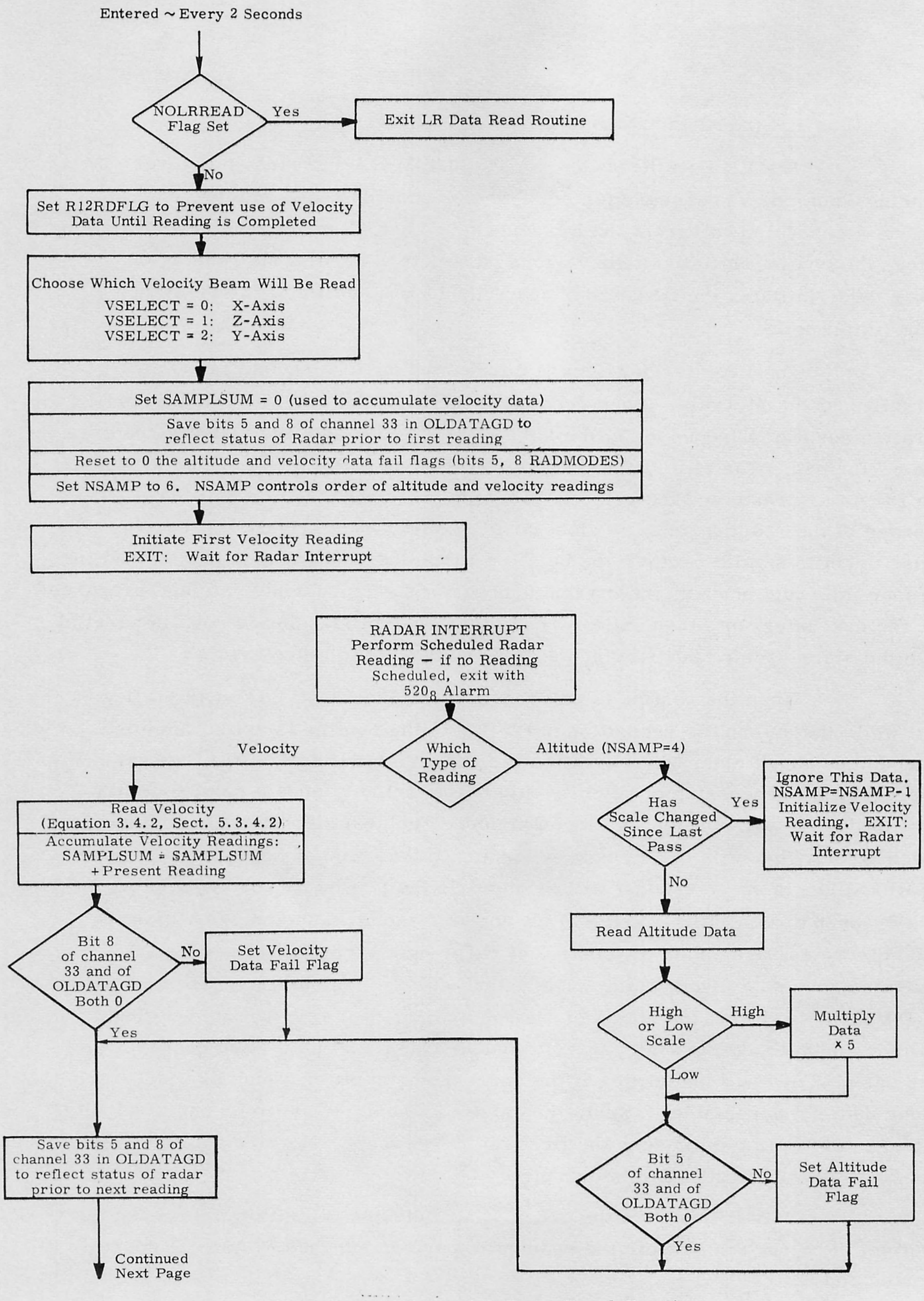


Figure 3.4.4-1. LR Date Read Routine

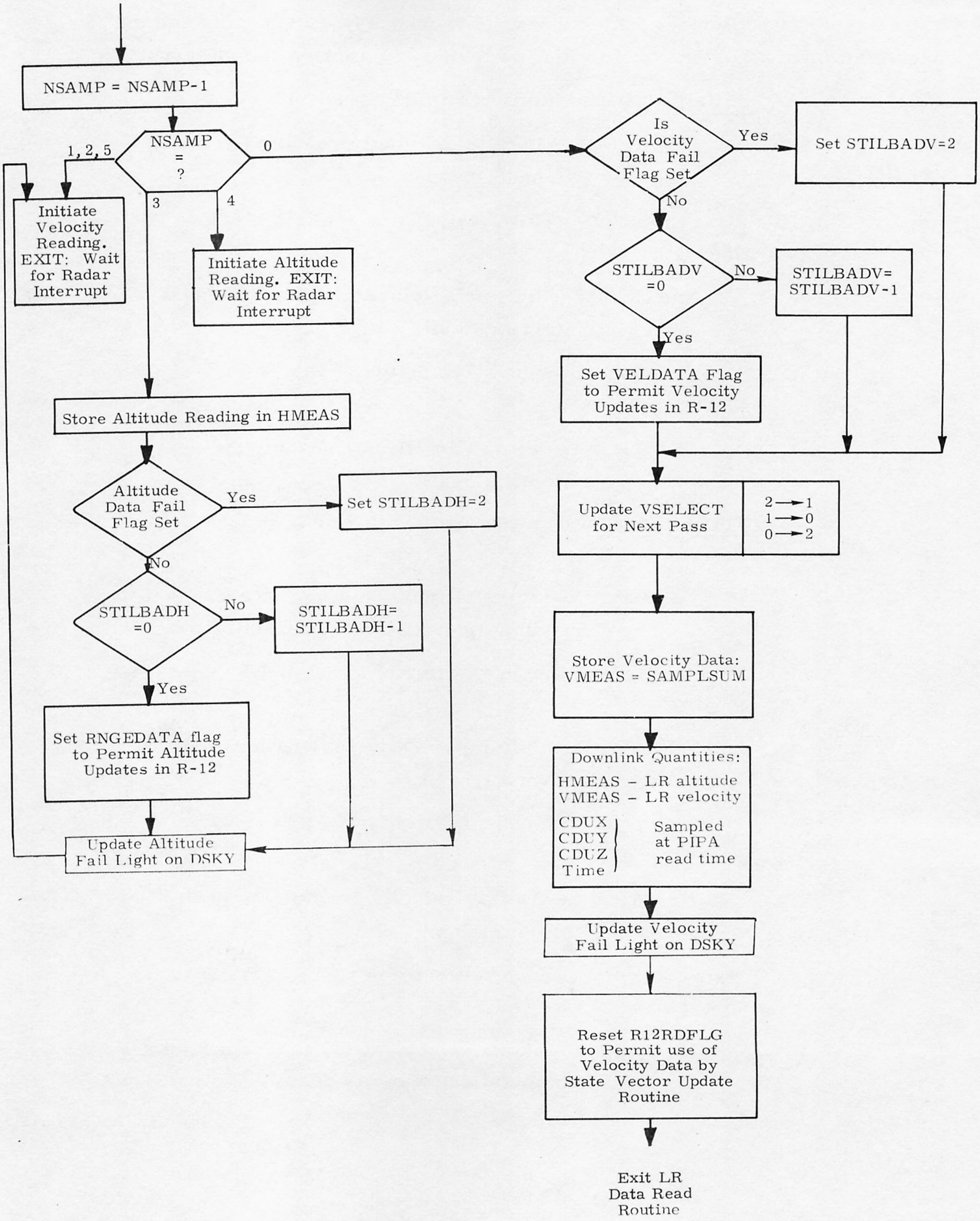


Figure 3.4.4-1 (Cont) LR Data Read Routine

The following table shows the sequence of interrupts, and the manner in which NSAMP controls the computation following the interrupt:

A. Initialization is entered, NSAMP is set to 6.

NSAMP = 6 — Initiate a velocity reading for next interrupt, exit.

B. First interrupt occurs, velocity is read, NSAMP = NSAMP - 1.

NSAMP = 5 — Initiate a velocity reading for next interrupt, exit.

C. Second interrupt occurs, velocity is read, NSAMP = NSAMP - 1.

NSAMP = 4 — Initiate an altitude reading for next interrupt, exit.

D. Third interrupt occurs, altitude is read, NSAMP = NSAMP - 1.

NSAMP = 3 — Store altitude data, initiate a velocity reading for next interrupt, exit.

E. Fourth interrupt occurs, velocity is read, NSAMP = NSAMP - 1.

NSAMP = 2 — Initiate velocity reading for next interrupt, exit.

F. Fifth interrupt occurs, velocity is read, NSAMP = NSAMP - 1.

NSAMP = 1 — Initiate velocity reading for next interrupt, exit.

G. Sixth interrupt occurs, velocity is read, NSAMP = NSAMP - 1.

NSAMP = 0 — Store velocity data, perform final computations for this pass.

When the radar interrupt occurs, the program checks to see which type of reading was scheduled (altitude or velocity). (If no reading was scheduled, a 520_8 alarm will occur, and the program will exit.)

For velocity reading, the data is read and accumulated in SAMPLSUM. Then the velocity data-good discrete (CHANNEL 33 bit 8) is checked to test the present status of the radar, and OLDATAGD bit 8 is checked for previous status. If either bit is 1, the velocity data fail flag (RADMODES, bit 8) is set to indicate bad data. At this point, NSAMP is decremented and checked, and depending on its value, the next interrupt is initiated, as outlined in the table above.

For an altitude reading, a check is first made as to whether the altitude scale factor has changed since the last pass. If this is the case, the data (probably bad) is ignored, and the next velocity reading is initiated. If the scale has not changed, the data is read. High-scale data is rescaled to low-scale format. The altitude data-good discrete is checked (CHANNEL 33 bit 5) as well as OLDATAGD, bit 5. If either bit is 1, the corresponding data fail flag is set (RADMODES, bit 5). Now NSAMP is decremented and checked.

When NSAMP = 3 (an altitude reading has just occurred) altitude data is stored in HMEAS for use by R12. The altitude data fail flag is then checked. If set (data bad), STILBADH is set to two. When clear, STILBADH is decremented (unless zero). STILBADH must be zero before setting the RNGEDATA flag (i. e., three consecutive passes with data good must have occurred). R12 will not use the altitude data until the RNGEDATA flag is set. Before exiting this branch, the next velocity reading is initiated.

When NSAMP = 0, the final velocity reading has been completed. The velocity data fail flag is checked to see if the data was bad for one or more of the five readings. In a manner similar to that described above, STILBADV is used to insure that the VELDATA flag will only be set after three consecutive passes with good velocity data. (R12 will not use the velocity reading until VELDATA is set.) VSELECT is now updated so that a new velocity beam will be read next pass. The accumulated velocity data (SAMPLSUM) is stored in VMEAS for use by R12. Finally, the R12RDFLG is reset to permit use of velocity data by R12.

5.3.4.5 Repositioning of LR Antenna

Logic flow for antenna repositioning is shown in Figure 3.4.5-1. When certain criteria have been met in the Descent State Vector Update Routine (R12), this logic is called to command the LR antenna from position 1 to position 2.

The reposition command is provided to the LR hardware via bit 13 of channel 12. When the radar repositions (or if 11 seconds have passed without repositioning) the command is removed and LRPOS is set to position 2. The NOLRREAD Flag is reset to permit use of radar data by R12.

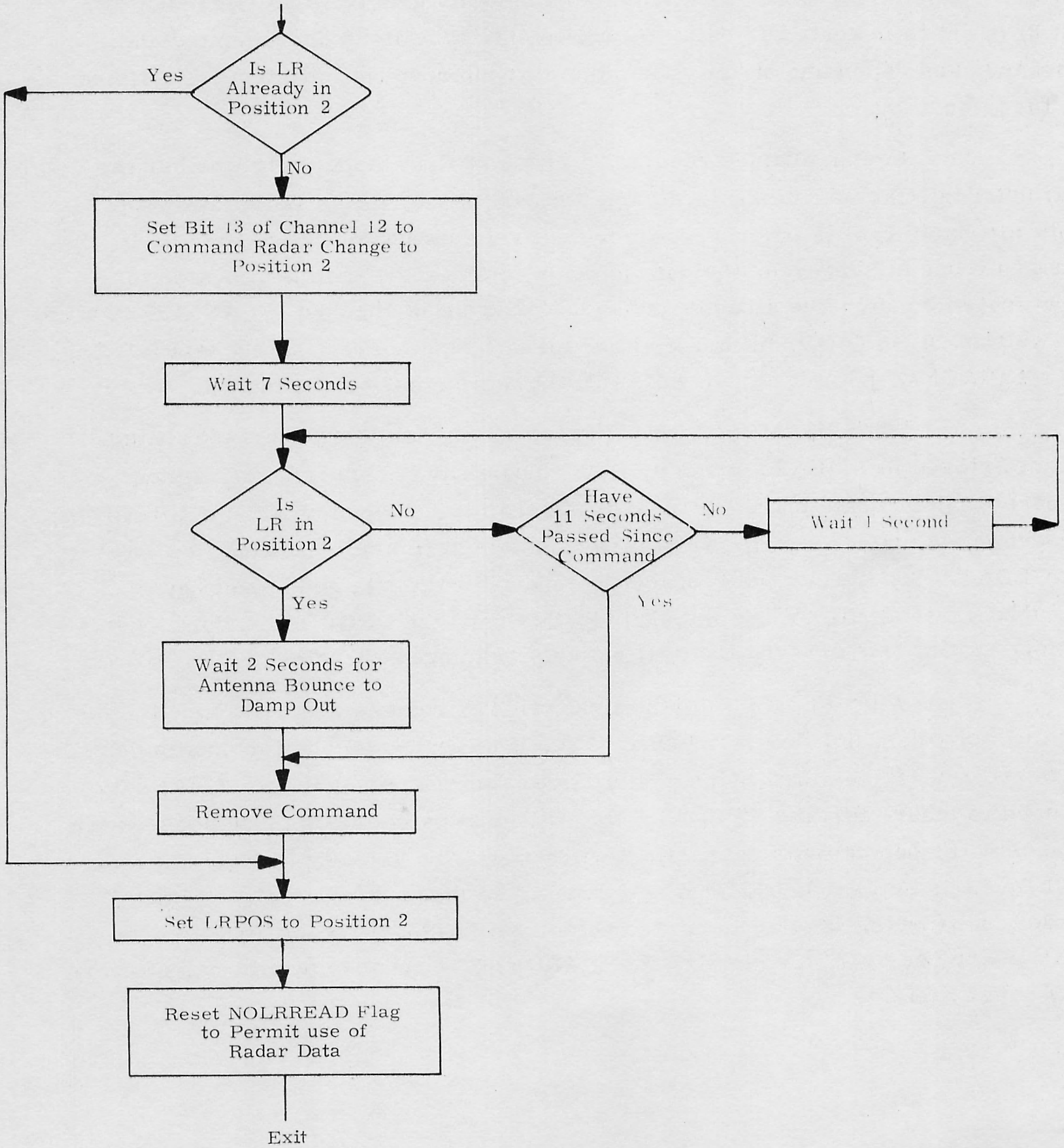


Figure 3.4.5-1. Repositioning of LR Antenna

5.3.4.6 Guidance-and-Control Routine

The Guidance-and-Control Routine uses the data provided by the State-Vector-Update Routine to produce a thrust acceleration command for the Throttle-Command Routine and desired thrust direction and window direction vectors for the FINDCDUW routine.

The Guidance-and-Control Routine is described by an overall flow chart, Figure 3.4.6-1, and by a detailed flow chart, Figure 3.4.6-2. Notes on interpreting the charts follow. The nomenclature used in the Guidance-and-Control section is defined at the end of this section.

In this document, the polarities of all variables, including padload-erasables, reflect the design concepts of the system. In certain cases, to facilitate LGC arithmetic, the LGC padloads are of opposite sign. To avoid cluttering the exposition of this section, LGC pecularities such as sign convention and scaling are not indicated. When opposite, the LGC sign convention is indicated in the Erasable Memory Parameter List, but scaling is beyond the scope of this document.

Notes on Interpreting Lunar Landing Guidance Information Flow

In this section, there are short paragraphs relevant to specific parts of the information flow diagram. These are identified by the notation (m, n), where m refers to the page of the information flow on which the relevant section may be found, and n refers to the particular note on that page. A corresponding identifier will be found to the right of the flow.

Small triangles at the bottom of the page containing integers only indicate that flow continues to the succeeding page at the triangle with the identical integer. Large triangles containing a letter followed by an integer indicate that control is passed to or from the large triangle with the identical letter on the page indicated by the integer.

Specific Notes:

(1, 1) a. REDFLAG, here initialized to zero (and again zeroed at the start of P64) to forbid redesignations, will be set to one only upon receipt of a PROCEED response to flashing display V06N64, while ΔT_{REDES} is non-zero. AZINCR1 and ELINCR1 are initialized to zero in response to this PROCEED . NOTERFLG and LRBYPASS initialized for State Vector Update Routine.

b. FLAUTOX is not shown initialized to zero because the selection of any new Major Mode causes it to be set to zero. (The State-Vector-Update Routine sets it to one when the proper altitude is passed.)

(1, 2) The ignition algorithm loop counter, n_1, prevents performing the pre-ignition computations endlessly without notice. It is difficult to imagine circumstances in which alarm code 1412, which n_1 controls, will be called.

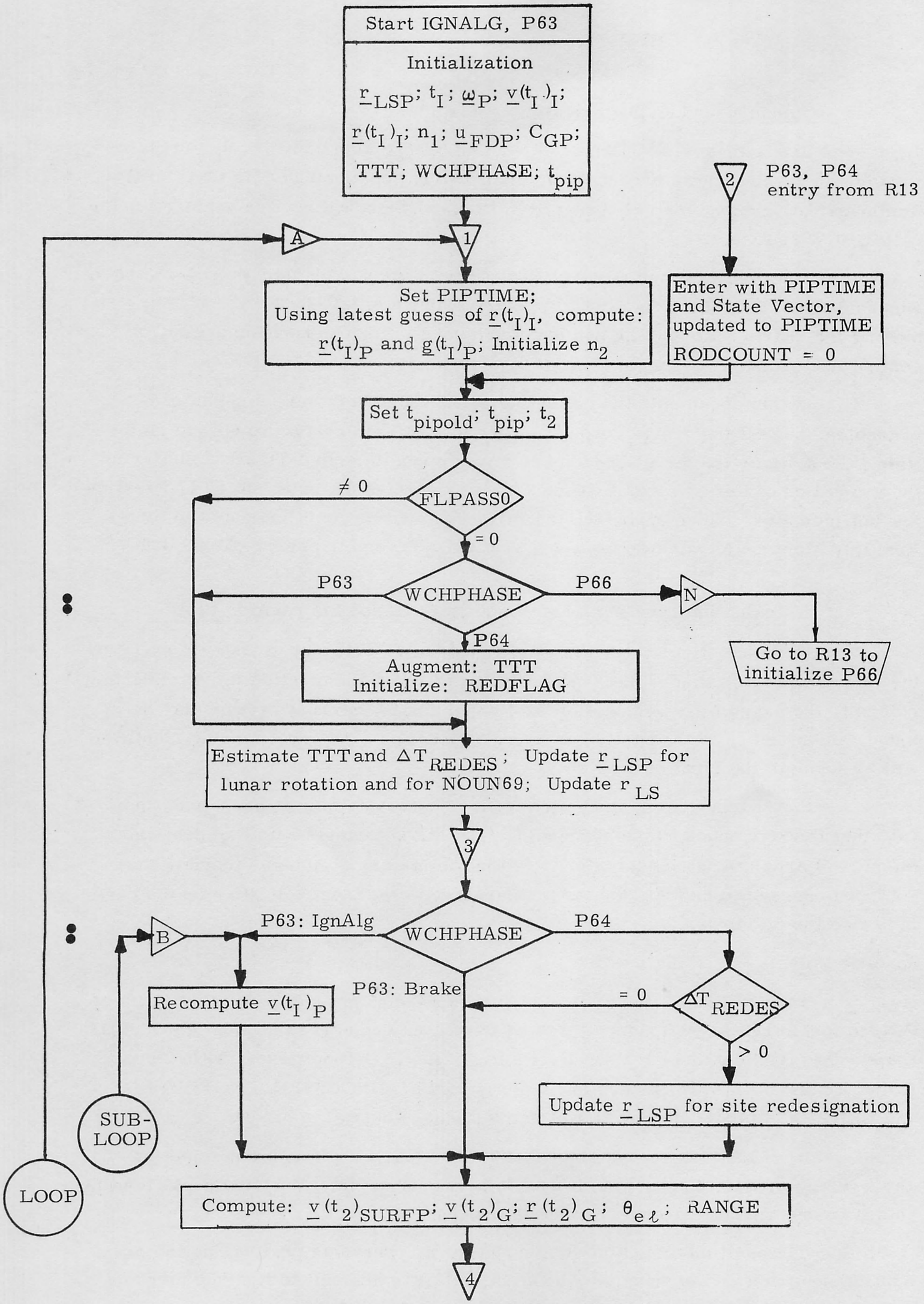
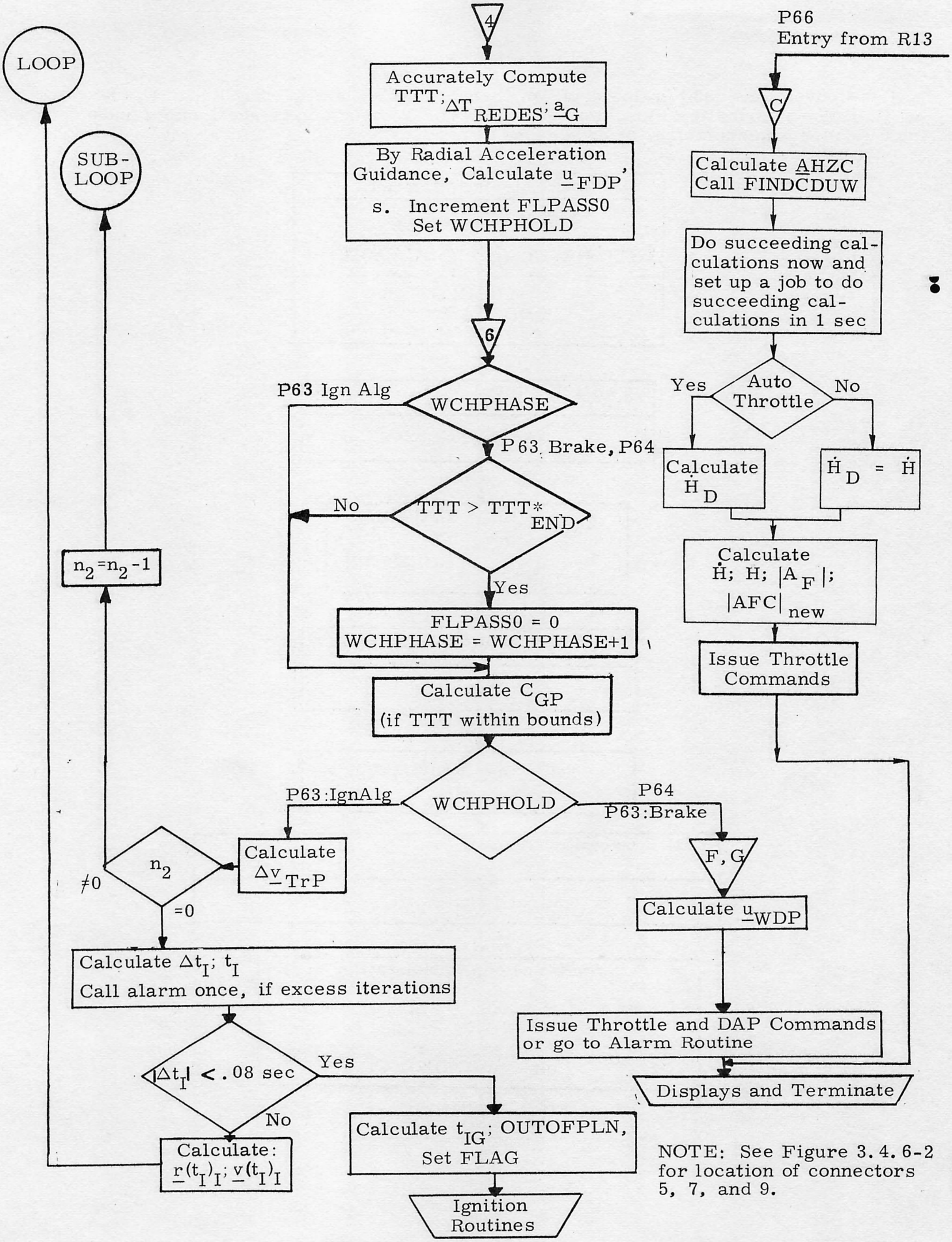


Fig. 3.4.6-1 Guidance-and-Control Routine - Overall Flow
Page 1 of 2



NOTE: See Figure 3.4.6-2 for location of connectors 5, 7, and 9.

Fig. 3.4.6-1 Guidance-and-Control Routine - Overall Flow
Page 2 of 2

For Conventions used in this diagram, see: Notes on Interpreting Lunar Landing Guidance Information Flow, this Section. For Nomenclature, see Lunar Landing Guidance Nomenclature, at end of diagram.

P63 LM: Initialization

Major Mode = P63

WCHPHASE = IgnAlg; FLPASS0 = 0

NOTERFLG = 0
LRBYPASS = 0
REDFLAG = 0 (1, 1)

↓

Call PIO: Input: $\underline{r}_{LSL}$, t_{LAND}

Output: $\underline{r}_{LSI}$

↓

$\underline{r}_{LSP}$ = REFSMMAT $\underline{r}_{LSI}$

$t_I = t_{LAND}$ - GUIDDURN

↓

Call PIO: Input: t_{comp}, (0, 0, 1)

Output: $\underline{\omega}_I$

↓

$\underline{\omega}_P = \omega_M$ REFSMMAT $\underline{\omega}_I$

$r_{LS} = |\underline{r}_{LSL}|$

↓

Call Integration: Input: t_I

Output: $\underline{r}(t_I)_I$, $\underline{v}(t_I)_I$

↓

Ignition Loop Initialization

$n_1 = 40$, $C_{GP} = I$, $\underline{u}_{FDP} = \Delta\underline{v}_{TrP} = (0, 0, 0)$ (1, 2)

TTT = 0; $t_{pip} = t_{LAND}$; $\Delta h = 99999$

↓

HLROFF = 50 ft.

↓

(1)

Fig. 3.4.6-2 Guidance-and-Control Routine
Page 1

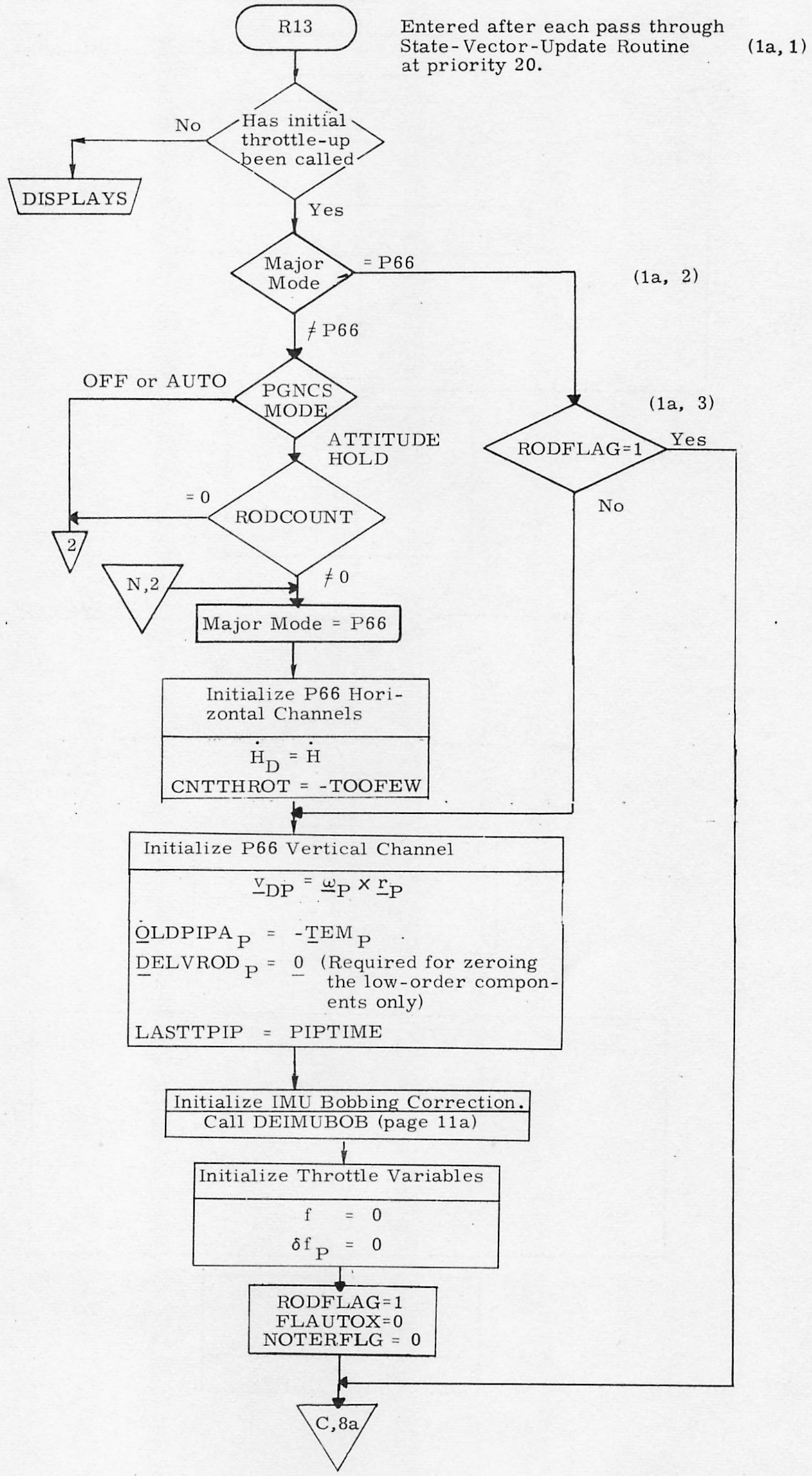


Figure 3.4.6-2. Guidance and Control Routine Page 1a

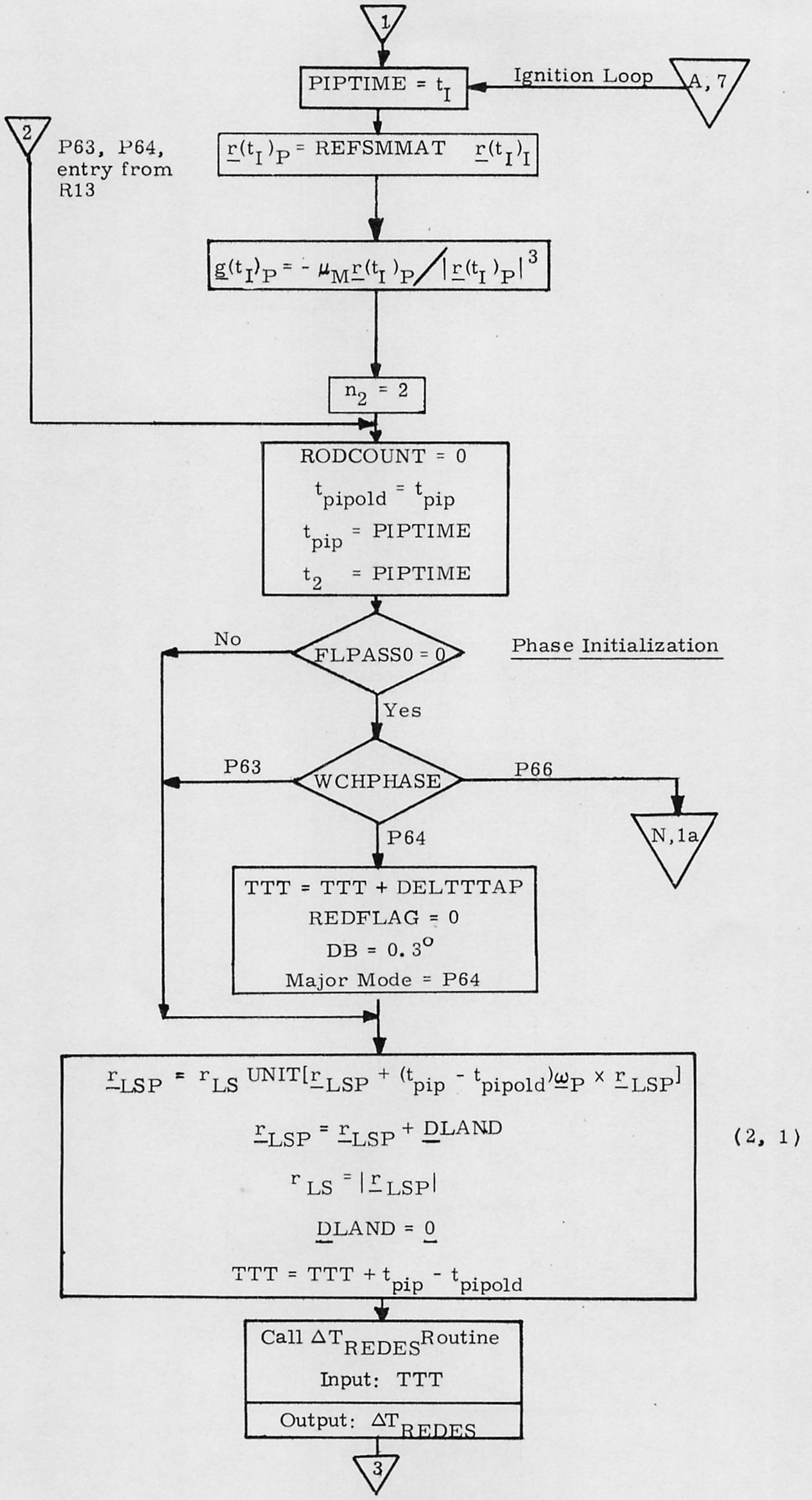


Figure 3.4.6-2 Guidance-and-Control Routine
Page 2

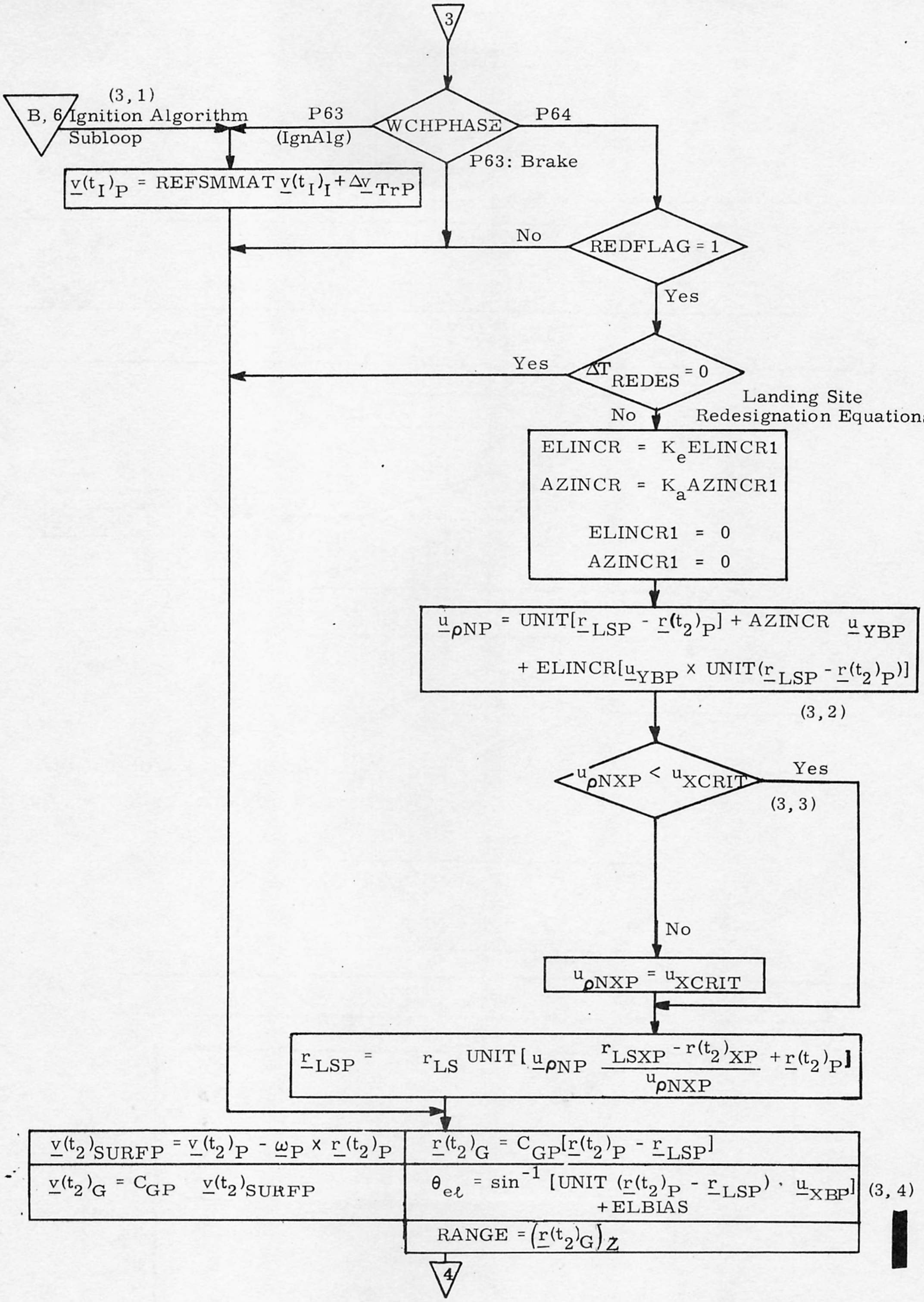


Fig. 3.4.6-2 Guidance-and-Control Routine

Page 3

4

$$COUNT_{TTT} = 0$$
$$\epsilon = |TTT/128|$$
$$TTT_{TEMP} = TTT$$

Compute Δ TTT to satisfy j_{DZG}

$$\Delta TTT = -\frac{j^*_{TZG}TTT^3 + 6a^*_{TZG}TTT^2 + [18v^*_{TZG} + 6v(t_2)_{ZG}]TTT + 24(r^*_{TZG} - r(t_2)_{ZG})}{3j^*_{TZG}TTT^2 + 12a^*_{TZG}TTT + 18v^*_{TZG} + 6v(t_2)_{ZG}}$$

$$TTT = TTT + \Delta TTT$$

$COUNT_{TTT} = 8$ — Yes: $TTT = TTT_{TEMP}$ → K, 11 (alarm routine)

No

$$COUNT_{TTT} = COUNT_{TTT} + 1$$

$|\Delta TTT| \leq \epsilon$ (4, 1) — No: return to Compute Δ TTT

Yes

Quadratic Acceleration Predictive Guidance Equations

$$TTT_P = TTT + LEADTIME$$
$$\text{If } TTT_P > 0,\ TTT_P = 0 \qquad (4,3)$$

Call ΔT_{REDES} Routine; Input: TTT; Output: ΔT_{REDES}

$$\begin{aligned} \underline{a}_G = & \left(-24\frac{TTT_P}{TTT^3} + 36\frac{TTT_P^2}{TTT^4}\right)\left(\underline{r}^*_{TG} - \underline{r}(t_2)_G\right) \\ & + \left(-18\frac{TTT_P}{TTT^2} + 24\frac{TTT_P^2}{TTT^3}\right)\underline{v}^*_{TG} \\ & + \left(-6\frac{TTT_P}{TTT^2} + 12\frac{TTT_P^2}{TTT^3}\right)\underline{v}(t_2)_G \\ & + \left(6\frac{TTT_P^2}{TTT^2} - 6\frac{TTT_P}{TTT} + 1\right)\underline{a}^*_{TG} \end{aligned} \qquad (4, 4)$$

5

Fig. 3.4.6-2 Guidance-and-Control Routine

Page 4

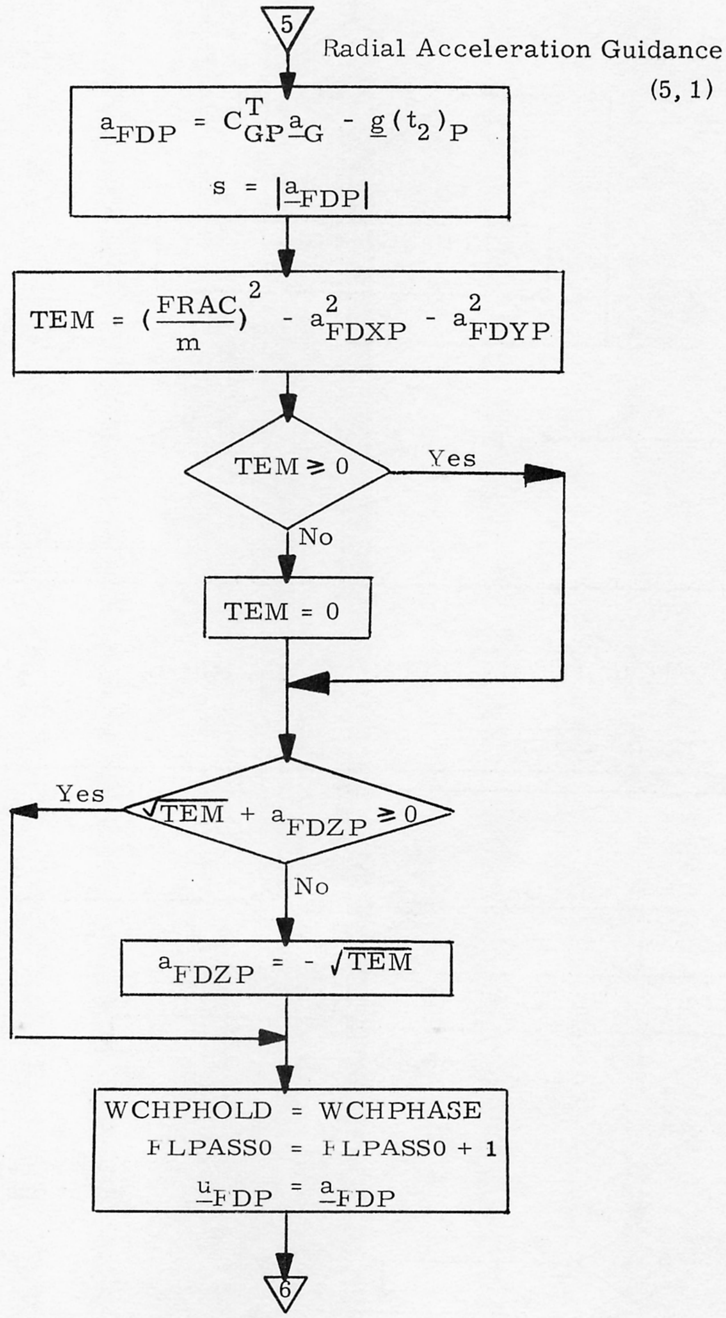


Fig. 3.4.6-2 Guidance-and-Control Routine
Page 5

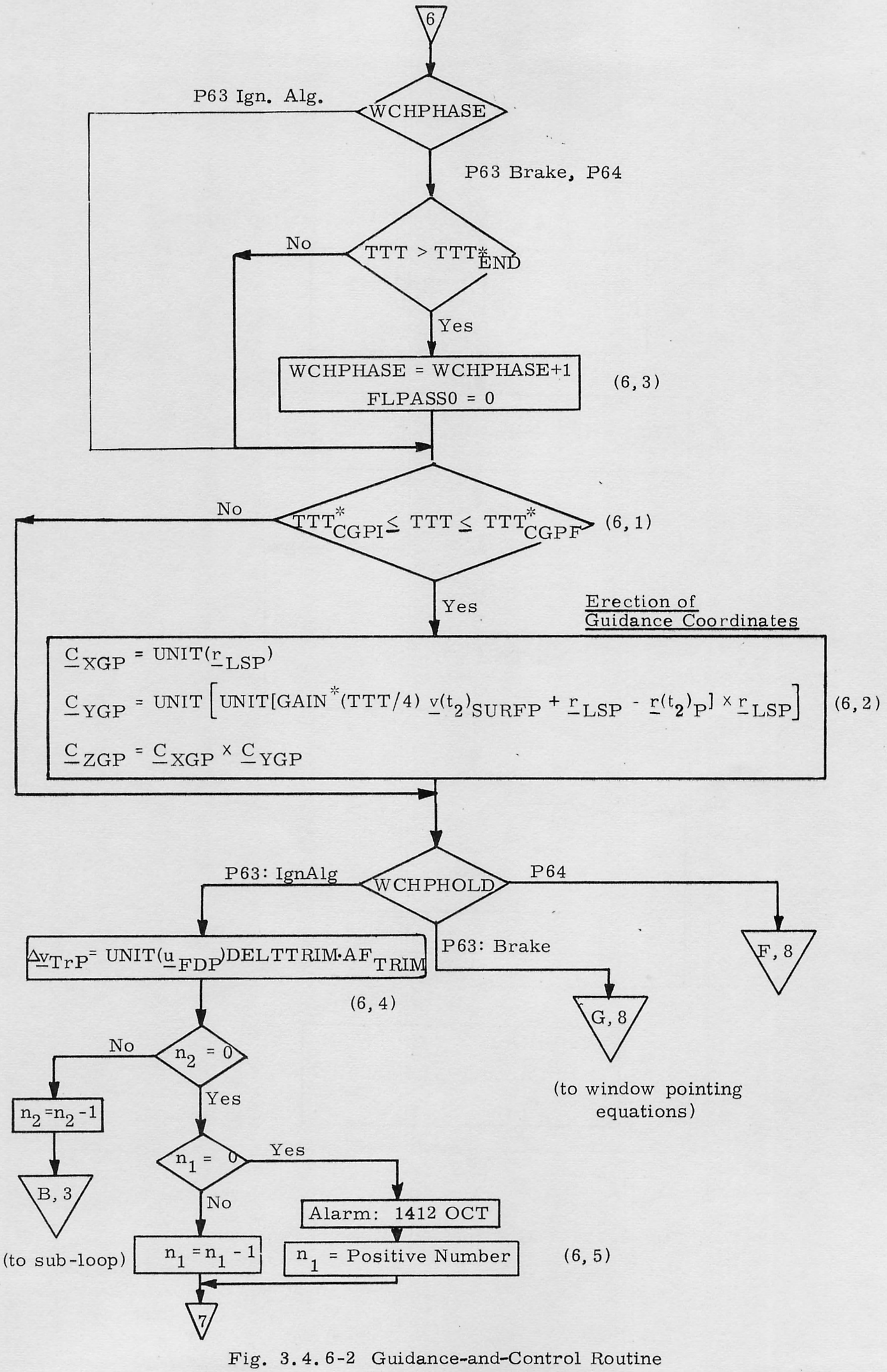


Fig. 3.4.6-2 Guidance-and-Control Routine

Page 6

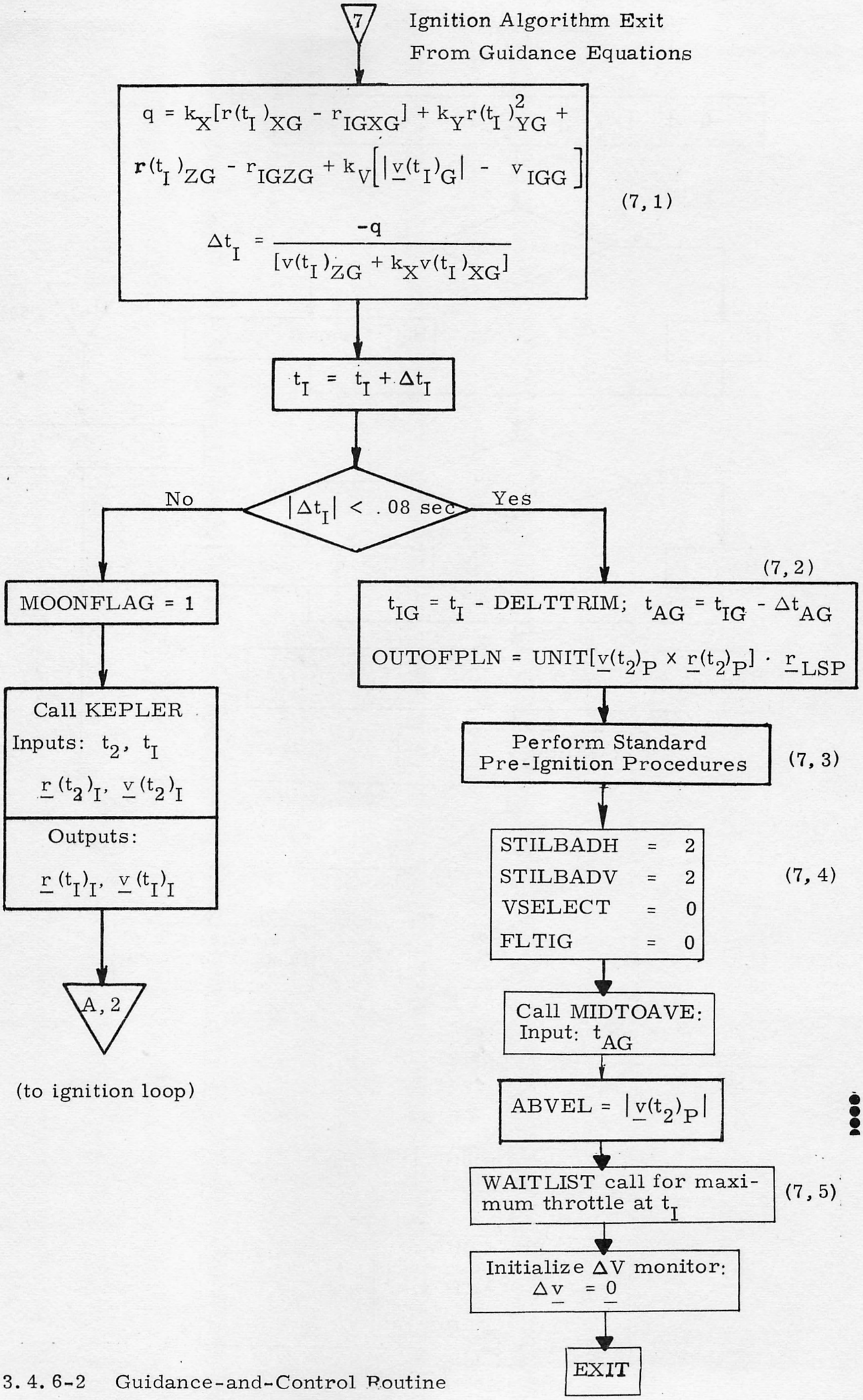


Figure 3.4.6-2 Guidance-and-Control Routine
Page 7

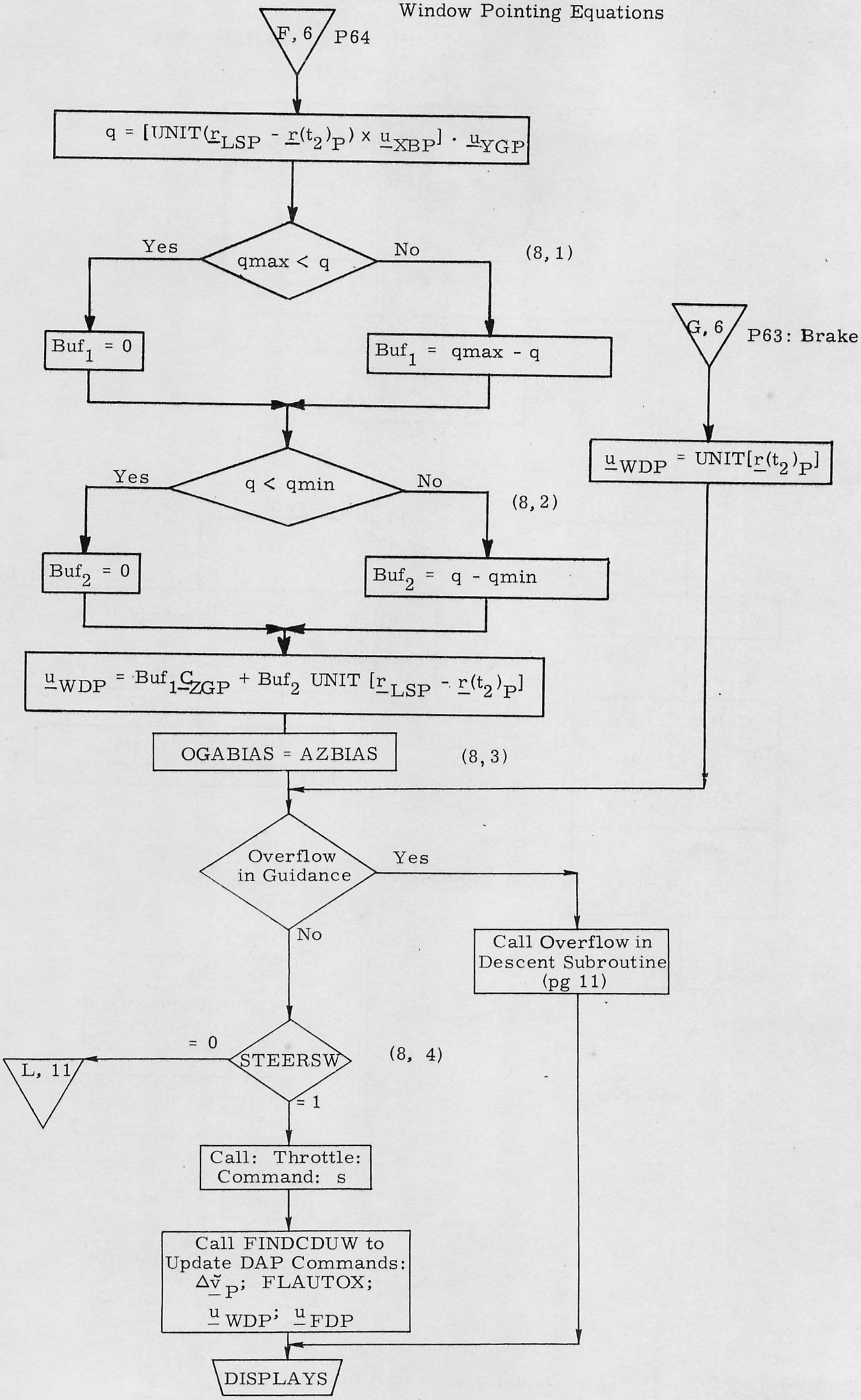


Fig. 3.4.6-2 Guidance-and-Control Routine
Page 8

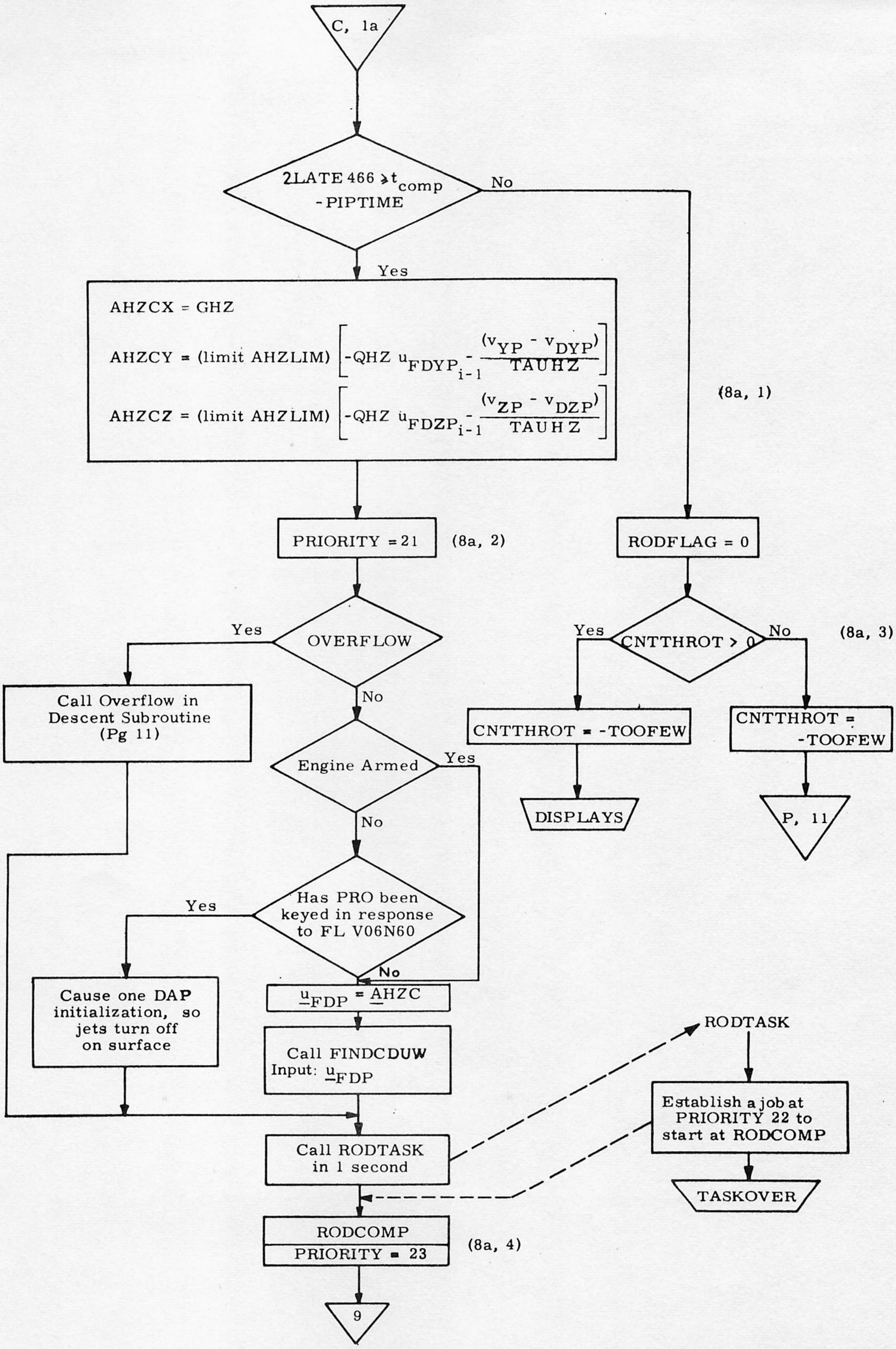


Fig. 3.4.6-2 Guidance-and-Control Routine (Page 8a)

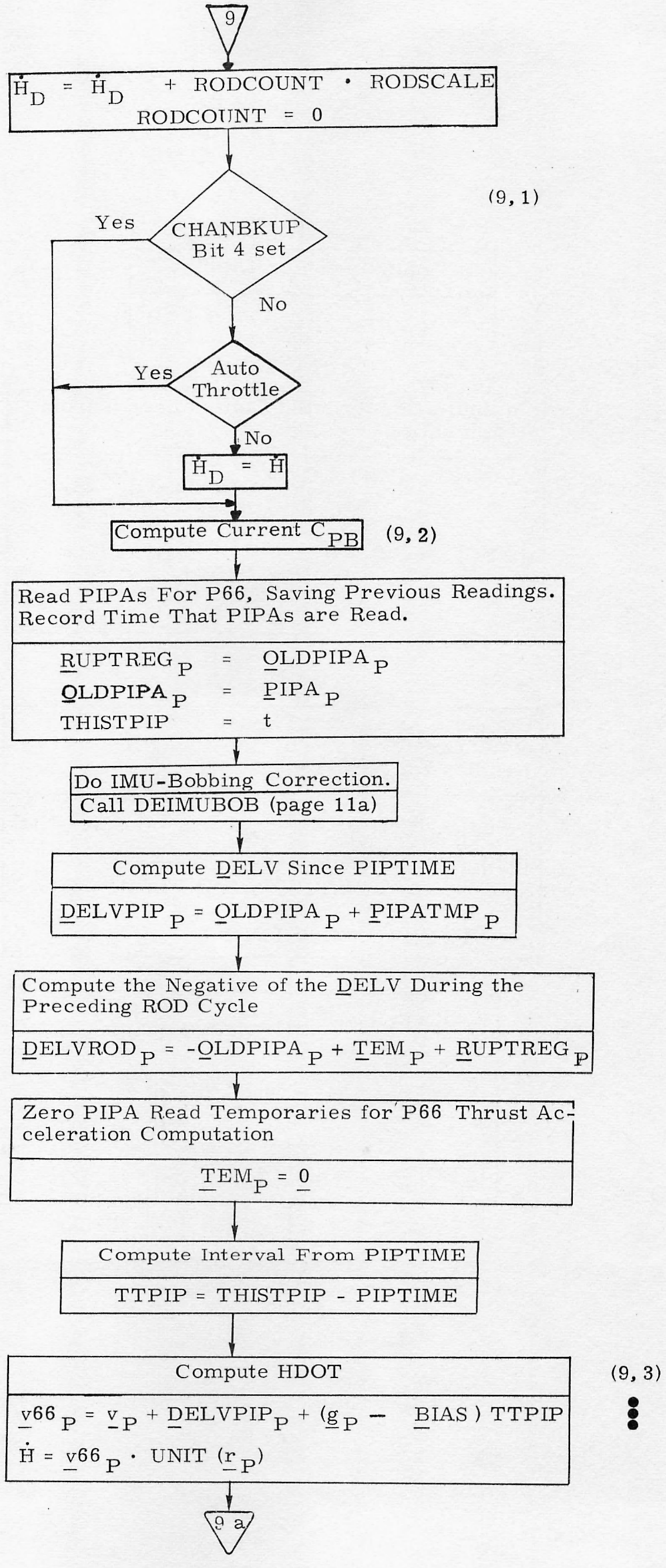


Figure 3.4.6-2. Guidance and Control Routine Page 9

9a

Compute H For Display

$$H = |\underline{r}_P| - r_{LS} + \dot{H}\ TTPIP$$

Compute Uncorrected Commanded Thrust Acceleration

$$\cos(\xi) = UNIT(\underline{r}_P) \cdot C_{PB}\,\underline{u}_{XB}$$

$$|\underline{A}FC|\ raw = \frac{\dfrac{\dot{H}_D - \dot{H}}{TAUROD} + |\underline{g}_P|}{\cos(\xi)}$$

Compute Measured Thrust Acceleration and Transfer Time of this P66 PIPA Reading for Next Pass.

$$|\underline{A}F| = \frac{|\underline{D}ELVROD_P + \underline{B}IAS \cdot 1\ SECOND|}{THISTPIP - LASTTPIP}$$

LASTTPIP = THISTPIP

$$|\underline{A}FC|_c = \frac{Lag}{TAUROD}\left[\frac{|\underline{g}_P|}{\cos(\xi)} - |\underline{A}F| - \frac{\delta f_P}{m}\right] \qquad (9a,\ 1)$$

$$|\underline{A}FC|_{new} = |\underline{A}FC|_c + |\underline{A}FC|_{raw}$$

10

Fig. 3.4.6-2 Guidance and Control Routine
Page 9a

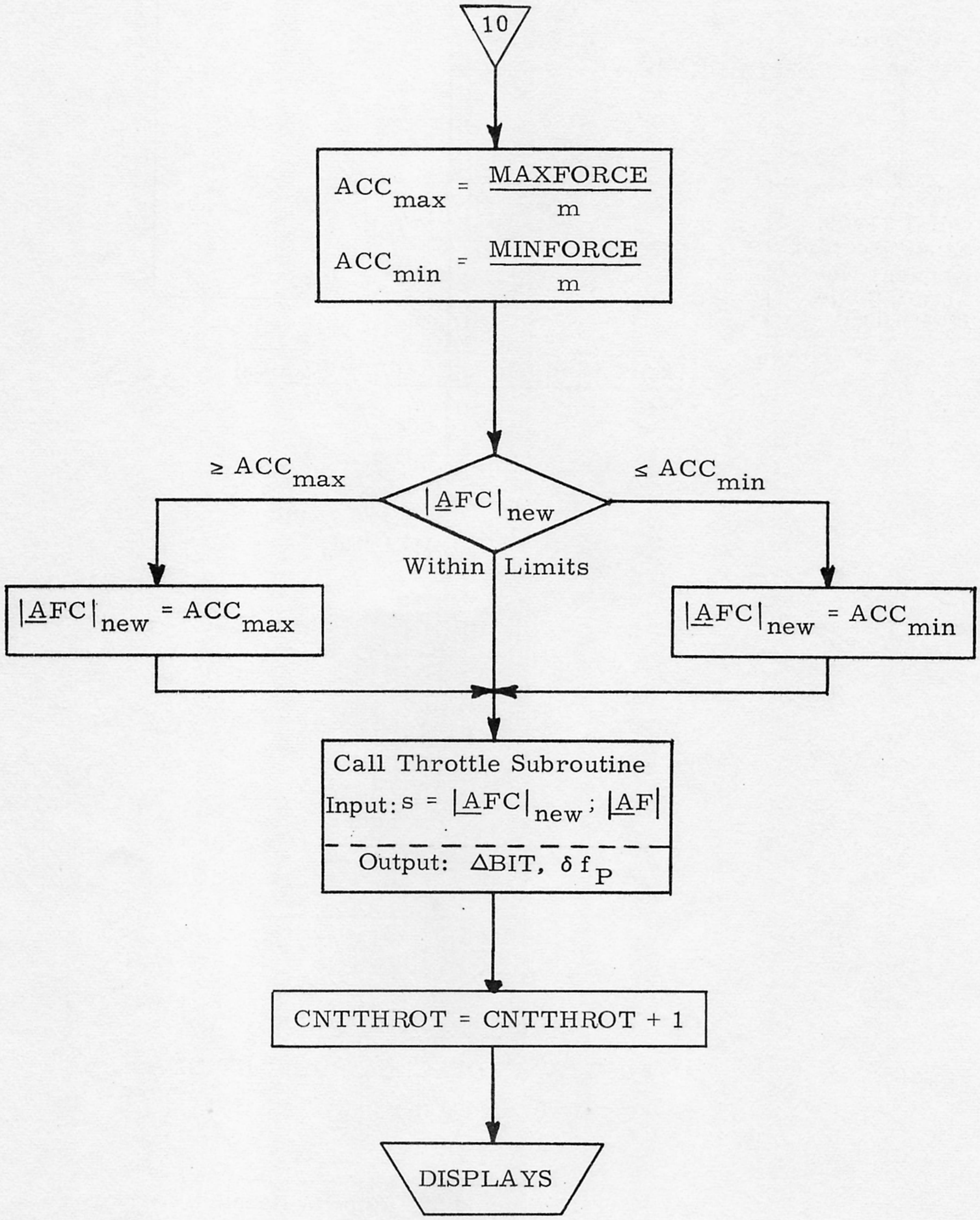
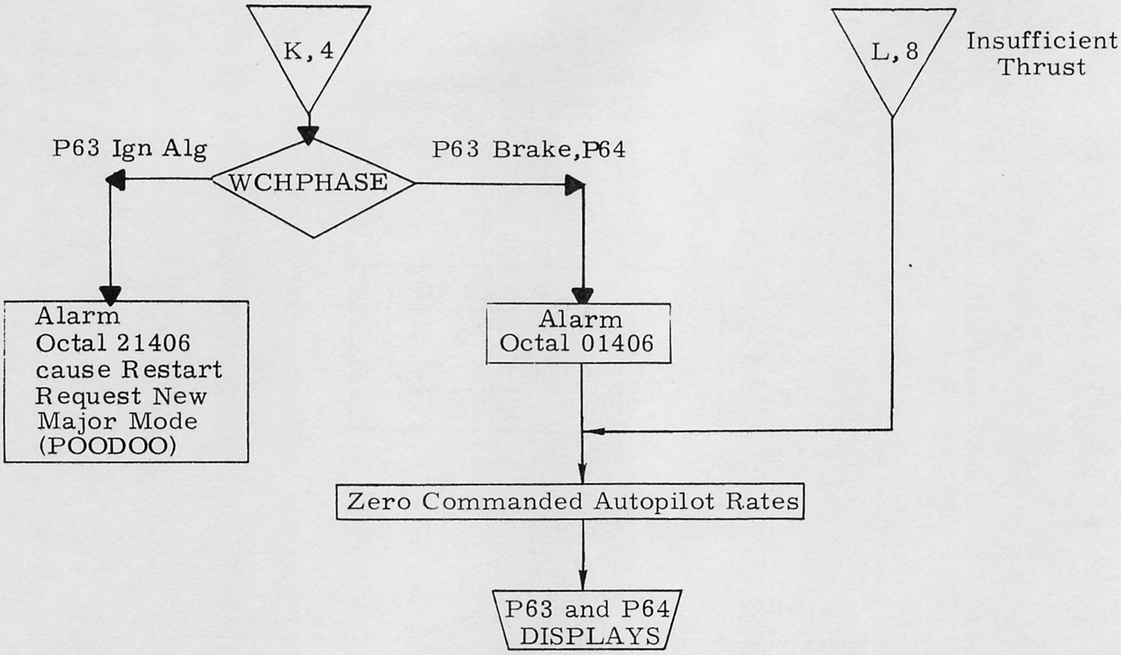
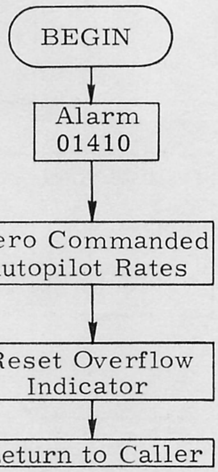


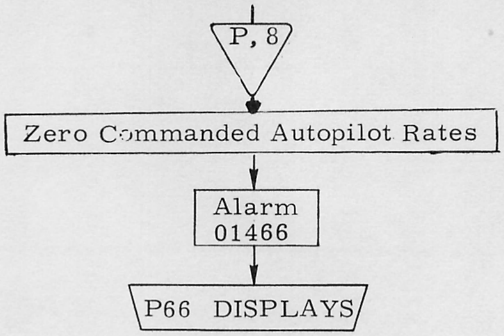
Fig. 3.4.6-2 Guidance-and-Control Routine
Page 10



TTT FAILURE

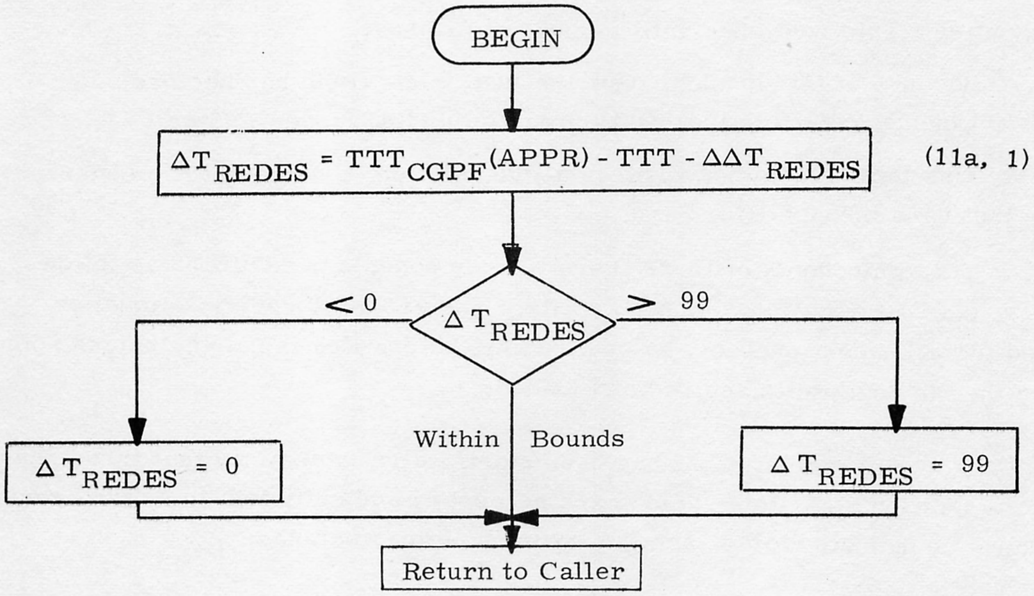


OVERFLOW IN DESCENT SUBROUTINE

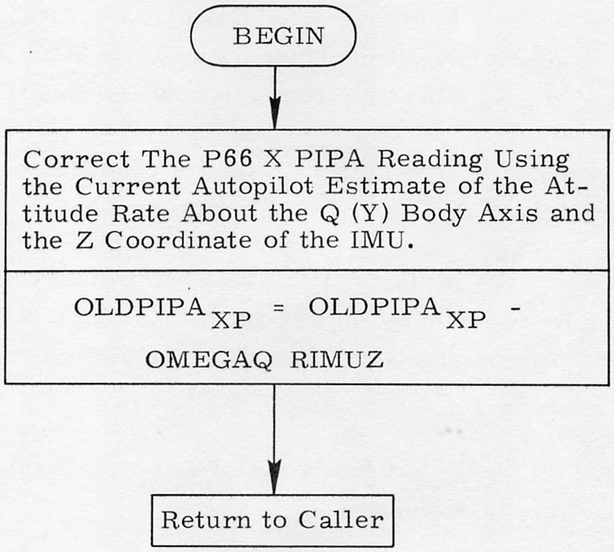


INSUFFICIENT THROTTLING IN P66

Lunar Landing Guidance Alarms
Figure 3.4.6-2 Guidance-and-Control Routine



ΔT_{REDES} ROUTINE



DEIMUBOB ROUTINE

Figure 3.4.6-2 Guidance and Control Routine
(page 11a)

(1a, 1) During each guidance cycle after the radar updates, if any, have been made, the State-Vector-Update Routine calls R13 which calls the Guidance-and-Control Routine. This continues until touchdown or abort.

(1a, 2) Because of this branch, once the ROD mode (P66) has been selected, the astronaut cannot return to the LGC automatic landing modes P63 or P64.

(1a, 3) This test determines whether a P66 deletion or a restart has occurred since the last pass through R13.

(2, 1) The components of $\underline{\text{DLAND}}$ are the components of NOUN 69 as follows: $DLAND_x = R_3$, $DLAND_y = R_2$, $DLAND_z = R_1$. The NOUN 69 registers must be initialized to zero (most probably by pad loading) before P63 is called, unless some change in the landing site is desired and loaded.

(3, 1) For a particular t_I, it has been empirically determined that three passes through the ignition algorithm subloop are necessary and sufficient to define accurately the guidance coordinate frame, and the thrust-pointing vector, $\underline{u}_{FDP}$.

(3, 2) Moving the Rotational Hand Controller forward (or left) produces a redesignated site forward (or left) of the previous landing site.

Thus, a + roll deflection creates a positive AZINCR,
a - roll deflection creates a negative AZINCR,
a + pitch deflection creates a negative ELINCR,
and a - pitch deflection creates a positive ELINCR.

(3, 3) The X-component of the LOS direction is checked at this point to make sure that the site has not been redesignated past the horizon.

(3, 4) $\theta_{e\ell}$ is computed by a polynomial which produces accurate results only in the region $\pm 70^o$. Because the window outline limits visibility short of this range, the error when $|\theta_{e\ell}| > 70^o$ is immaterial. The 70^o restriction mentioned here should exclude the effect of ELBIAS.

(4, 1) This test on ΔTTT is usually satisfied on the first iteration cycle, except for the first guidance cycle of a given phase.

(4,3) TTT_P is used in the calculation of $\underline{a}_G$ to account for the lag in the guidance system between the time when the PIPAs are read (for a particular cycle) and the time when the corresponding commands to the throttle and DAP are actually implemented, thereby avoiding oscillations due to that lag.

(4,4) The target vector Y-components are forced to be zero.

(5,1) Radial acceleration guidance greatly reduces trajectory altitude variations due to engine dispersions.

(6,1) During the Ignition Algorithm, this test on TTT ensures that the guidance frame will not be erected if TTT has a value such that the frame might be reversed. This test should not fail after the first pass through the ignition algorithm. See also Note (11a, 1).

(6,2) The internal unitization in the computation of C_{YGP} is done to keep significance in the fixed-point arithmetic of the LGC, even though it is analytically superfluous.

(6,3) FLPASS0 is set to zero to ensure that phase initialization, shown on page 2 of the figure, will take place on the succeeding pass.

(6,4) $\Delta\underline{v}_{TrP}$ is computed because, while the trim phase does not alter $\underline{r}(t_I)_P$ appreciably, it does change $\underline{v}(t_I)_P$ significantly.

(6,5) If the ignition algorithm alarm is issued, a peculiarity of the LGC coding causes n_1 to be reset to the integer whose octal value is the GENADR* of the LGC location in which the alarm code is stored. This number depends upon the LGC assembly revision number, so it cannot be predicted prior to program release. However, it is safe to assume that this number will generally be sufficiently large to prevent the alarm from being issued more than once.

(7,1) The computation of Δt_I is made because it is very unlikely that actual conditions at t_I will be sufficiently close to nominal to yield nominal throttle control duration. Since the trajectory before ignition cannot be changed, the ignition algorithm

*See AGC4 MEMO#9 – Block II Instructions, by Hugh Blair-Smith, MIT/IL, Revised 1 June 1967.

compensates for pre-ignition trajectory dispersions by adjusting the time at which maximum throttle will be commanded, t_I. For example, if the altitude at t_I is too great, Δt_I will be negative, allowing more time for braking. All relationships are linear, except when the craft is out of plane (non-zero Y-component of position in guidance coordinates), in which case the relationship is quadratic.

(7, 2) The computation of t_{AG} is shown at this point to provide a logical interface with the GSOP description of the MIDTOAVE Routine. In the LGC, t_{AG} is computed by the Master Ignition Routine.

(7, 3) Among the so-called Standard Pre-Ignition Procedures is the Attitude Maneuver Routine (R60) using the vector $\underline{u}_{FDP}$ from the ignition algorithm. It should be noted that the computed navigation state vector at the end of the ignition algorithm $(\underline{r}(t_2)_P, \underline{v}(t_2)_P)$ is not the correct initial state vector for the State-Vector-Update Routine, since the guidance cycle is started before t_{IG}.

(7, 4) These parameters are used by State Vector Update and MIDTOAVE routines.

(7, 5) This WAITLIST call is issued by the LGC Master Ignition Routine. It causes the Throttle-Command Routine to be entered at the location FLATOUT at time t_I. t_I can be slipped by R41.

(8, 1) If q is greater than qmax, then the landing site is visible to the astronaut and the window is pointed along the line of sight.

(8, 2) If q is less than qmin, then the landing site is far below the bottom of the window edge, and the window is simply pointed in the direction of the Z-axis of the guidance frame.

(8, 3) OGABIAS is zero for all thrusting programs except the lunar landing.

(8, 4) STEERSW = 0 means insufficient thrust.

(8a, 1) The notation (limit AHZLIM) indicates that the content of the subsequent brackets is magnitude limited to AHZLIM. The notation i-1 indicates the use of the command from the previous pass, and on the first P66 pass indicates the command from the final P64 pass.

(8a, 2) The priority structure of AGC jobs is not ordinarily treated in this volume. It is discussed here, however, because of its extraordinary significance to the landing guidance. The effect of this priority structure is to allow only one P66 job to run at one time. The change to priority 21 is shown in this location in order to avoid breaking up the preceding equations. Actually, AHZCX is set to GHZ, and AHZCY and AHZCZ are magnitude limited, after the priority change.

(8a, 3) If too few calls to the Throttle-Command Routine are made between P66 deletions, a non-abortive alarm is issued.

(8a, 4) Since the ROD cycle time is one second, a WAITLIST task schedules ROD at this point every other second.

(9, 1) The astronaut can bypass the auto throttle check (channel 30 bit 5) by setting CHANBKUP bit 4 to 1.

(9, 2) There is no LGC equivalent of the MATRIX C_{PB}. In the LGC a vector is transformed from platform to body coordinates and vice versa by the LGC subroutine *SMNB* and *NBSM* by rotating the vector through each successive gimbal angle, never explicitly calculating a transformation matrix.

(9, 3) BIAS is the delta v bias accruing per second, and thus has the units of acceleration.

(9a, 1) $|\underline{A}FC|_c$ corrects $|\underline{A}FC|_{raw}$ for an extrapolation of $\dot{H}$ through the time interval Lag. At the extrapolated time, the change in thrust is expected to be, in effect, achieved. This extrapolation prevents overshoots in $\dot{H}$ in response to step changes in commanded $\dot{H}$. See MIT/DL publication E-2543.

(11a, 1) The constant $\Delta\Delta T_{REDES}$ assures one, and probably two, guidance frame erections after the final site redesignation. See also Note (6, 1).

5.3.4.6.1 Nomenclature of the Guidance-and Control Routine

Conventions

1. A superscript asterisk denotes a phase-dependent quantity; i.e., one erasable is indicated for the braking phase, a different erasable for the approach phase. If one quantity is always used regardless of phase, as in the computation of ΔT_{REDES}, the phase is indicated by a parenthetical note.

2.a. Time quantities measured relative to liftoff are denoted by a name starting with lower-case t.

b. Time quantities measured relative to the time the spacecraft will (or if allowed would) achieve the projected target conditions are denoted by a name starting with TTT.

c. Quantities representing an interval of time are denoted by a name starting with Δt.

3. Names of vector quantities are, in general, derived from the following building blocks:

Δ	Vector denotes change
$\underline{u}$	Unit vector
$\underline{r}$	Position vector
$\underline{v}$	Velocity vector
$\underline{a}$	Acceleration vector
$\underline{j}$	Jerk vector $(d^3\underline{r}/dt^3)$
( )	Time of vector validity in parentheses

Random Modifiers:

LS	Landing Site
T	Target
D	Desired

Coordinate Frames, always the last letter in the subscript:

G	Guidance
P	Platform
L	Lunar-Fixed (Selenographic)
I	Inertial (Basic Reference)

4. Components of a particular vector are denoted by the name of the vector, with no letters underlined, and with X, Y, or Z inserted in the penultimate position of the subscript.

5. Transformation matrices, with the exception of REFSMMAT, are denoted by C plus two subscripts, the first of which indicates the frame of the vector after transformation, and the second of which denotes the frame of the vector before transformation by the matrix.

For example, C_{GP} will transform a vector $\underline{v}$ to guidance from platform coordinates in this manner:

$$\underline{v}_G = C_{GP}\underline{v}_P$$

6. Row vectors of a matrix are denoted as follows:

$$C_{GP} = \begin{bmatrix} \underline{c}_{XGP} \\ \underline{c}_{YGP} \\ \underline{c}_{ZGP} \end{bmatrix}$$

7. Letters F and P under the heading TYPE indicate the type of storage used for constants which are Fixed and Padloaded-Erasable, respectively. V indicates variable.

TIME QUANTITIES

NAME	TYPE	DEFINITION
DELTTRIM	P	Duration of trim maneuver.
DELTTTAP	P	An augment added to TTT during initialization for the first guidance cycle of the approach phase. When the value of TTT computed on the final guidance cycle of the braking phase is thus augmented, it nominally becomes the correct value for the first guidance cycle of the approach phase. This causes the subsequent iterative computation of TTT to converge on the first or second pass. Without this augment, the computation of TTT would require many iterations on the first guidance cycle of the approach phase and would needlessly waste computer time.
ΔT_{REDES}	V	Time remaining to redesignate the landing site.
$\Delta\Delta T_{REDES}$	F	Time interval to assure at least one computation of guidance coordinates after the last site redesignation.
GUIDDURN	F	Estimated time interval from full throttle time to t_{LAND}.
LEADTIME	P	A positive time increment used to compensate for the computation delay, plus the average of the delay in achieving the throttle change and the delay in achieving the attitude change.
PIPTIME	V	Time of validity of the state vector provided either by the pre-ignition computations or by the State-Vector-Update Routine.
t_{comp}	V	Time at which a computation takes place. That is, t_{comp} is the current time, as taken from the computer clock.
$t_{compold}$	V	Previous t_{comp}.

NAME	TYPE	DEFINITION
Δt_{comp}	V	$t_{comp} - t_{compold}$.
t_I	V	The time computed by the pre-ignition computations for commanding maximum throttle (start of braking phase).
Δt_I	V	Correction to the maximum throttle time at the start of the braking phase to correct for deviations from nominal of the state at that point. More precisely, Δt_I corrects for the residual in the equation relating the various components of the state vector deviation from the nominal at the t_I computed on the current iteration of the pre-ignition computations.
t_{IG}	V	Time of DPS ignition (the start of the TRIM phase, minimum throttle).
t_{LAND}	P	Time of landing (initial estimate).
t_{pip}	V	A storage location for PIPTIME used by descent guidance.
t_2	V	A short synonym for t_{pip}.
t_{pipold}	V	Contains t_{pip} of the previous pass through descent guidance.
TTT	V	Time relative to achieving target conditions, a negative number.
TTT^*_{CGPF}	P	The value of TTT at which computation of the matrix C_{GP} must be terminated.
TTT^*_{CGPI}	P	The value of TTT at which the computation of the matrix C_{GP} can be initiated.
TTT^*_{END}	P	Criterion for phase switching.
TTT_P	V	A predicted value of TTT to account for the total delay between reading the accelerometers and achieving the thrust acceleration commanded on the basis of those accelerometer readings.
TTT_{TEMP}	V	Temporary quantity for TTT.

Names of VECTOR QUANTITIES not derived as in part 3 of Nomenclature

NAME	TYPE	DEFINITION
$\Delta\underline{v}_{TrP}$	V	Nominal velocity increment accrued due to the thrust acceleration during the trim phase.
$\underline{D}LAND$	V	Change in the landing site vector requested via N69. Must be initialized to zero by pad loading.
$\underline{g}(t_2)_P$	V	Acceleration due to gravity at time t_2.
$\underline{\omega}_P$	V	Lunar rotational angular velocity, platform coordinates.
$\underline{u}_{FDP}$	V	Thrust acceration command vector, platform coordinates. This vector is given a name usually associated with unit vectors because it is an output from the Guidance-and-Control Routine to the FINDCDUW Routine and to the Attitude Maneuver Routine which is used as a <u>pointing</u> Command only. There is no requirement for this vector to be of unit (or semi-unit) length, and it is not unitized.
$\underline{u}_{\rho NP}$	V	Unit vector from the current LM position to the redesignated landing site, platform coordinates.
$\underline{u}_{WDP}$	V	Window-pointing vector, not necessarily unit. See definition of $\underline{u}_{FDP}$.
$\underline{v}(t_2)_{SURFP}$	V	Velocity with respect to the surface, in platform coordinates.

MISCELLANEOUS QUANTITIES

NAME	TYPE	DEFINITION
AF_{TRIM}	F	Thrust acceleration expected during Trim Phase.
AZBIAS	P	Azimuth bias to correct LPD alignment error. See also ELBIAS and OGABIAS.
AZINCR	V	Commanded azimuth angle change accrued since last pass through guidance, radians.
AZINCR1	V	Net count of AZIMUTH redesignation commands issued since last pass through Guidance-and-Control Routine.
DB	V	DAP Deadband width.
ELBIAS	P	Elevation bias to correct LPD alignment error. See also AZBIAS and OGABIAS.
ELINCR	V	Commanded elevation angle change accrued since last pass through guidance, radians.
ELINCR1	V	Net count of elevation redesignation commands issued since last pass through Guidance-and-Control Routine.
FLPASS0	V	A flag which, when zero, indicates the initial guidance cycle of any given phase of the lunar landing.
FRAC	F	Maximum available thrust from DPS (used for radial control guidance).
GAIN*	P	Gain factor, to reduce excessive rotation of the guidance coordinate frame, especially during site redesignation, thereby also reducing excessive yawing during approach phase.
K_a, K_e	F	Scale factors relating AZINCR and ELINCR to AZINCR1 and ELINCR1.
k_V, k_X, k_Y	P	Coefficients used by the pre-ignition computations in computing Δt_I.
m	(P)	Mass of the vehicle: register is pad-loaded, although the astronaut has the option to reload it after R03 is begun. The mass is updated by State Vector Update Routine.
n_1	V	Ignition algorithm loop counter.

NAME	TYPE	DEFINITION
n_2	V	Ignition algorithm subloop counter.
OGABIAS	P	Outer gimbal angle bias input to FINDCDUW. See also AZBIAS and ELBIAS.
OUTOFPLN	V	Amount landing site is out of CSM orbital plane.
qmax, qmin	F	Upper and lower criteria used in choosing window-pointing command vector.
r_{IGZG}, r_{IGXG}	P	Components of the nominal position vector in guidance coordinates at t_{LAND} - GUIDDURN
s	V	Specific force, or thrust acceleration.
u_{XCRIT}	F	Maximum value for elevation increment. Ensures site will not be redesignated above the horizontal plane, which would yield a site to the rear of the current LM position.
v_{IGG}	P	Absolute value of nominal velocity in guidance coordinates at t_{LAND} - GUIDDURN.
WCHPHASE	V	The current phase of the lunar landing.
ω_M	F	Rotational rate of the moon with respect to inertial space (radians/CS).

FLAGS

NAME	MEANING WHEN = 1	MEANING WHEN = 0
FLAUTOX	Inhibit X-axis override	Allow X-axis override
REDFLAG	Landing site redesignation permitted	Landing site redesignation not permitted
RODFLAG	No P66 deletion or Restart has occurred since RODFLAG last set	P66 deletion or restart has occurred
STEERSW	Sufficient thrust - Call Throttle-Command Routine and FINDCDUW Routine	Insufficient thrust - Do not call Throttle-Command Routine or FINDCDUW Routine

NAME	TYPE	DEFINITION
ACC_{max}, ACC_{min}	V	Current limits on thrust acceleration.
$\lvert \underline{A}F \rvert$	V	Magnitude of the measured thrust acceleration vector during interval since last pass through ROD computations.
$\lvert \underline{A}FC \rvert_c$	V	Magnitude of the commanded thrust acceleration vector correction for the computation and engine time delays.
$\lvert \underline{A}FC \rvert_{new}$	V	Magnitude of the commanded thrust acceleration vector corrected and issued to the Throttle-Command Routine.
$\lvert \underline{A}FC \rvert_{raw}$	V	Magnitude of the commanded thrust acceleration vector based on the altitude rate error at the P66 PIPA read time.
$\underline{A}HZC$	V	Commanded thrust acceleration vector for nulling horizontal velocity error.
AHZCX, AHZCY, AHZCZ	V	Components of $\underline{A}HZC$
AHZLIM	P	Maximum allowable value of AHZCY and AHZCZ
$\underline{B}IAS$	P	PIPA bias acceleration.
CNTTHROT	V	Counts number of times the Throttle-Command Routine has been called since last initialization of CNTTHROT. Initialized at -TOOFEW.
$\underline{D}ELVPIP_P$	V	Total PIPA count since PIPTIME.
$\underline{D}ELVROD_P$	V	The negative of the total PIPA count since last pass through ROD computations.
δf_P	V	F-weight: Commanded force times total system lag divided by the guidance cycle interval.
GHZ	F	Approximate Lunar Gravity at P66 altitude.
$\underline{g}_P$	V	Acceleration due to gravity at PIPTIME.
H	V	Altitude
$\dot{H}$	V	Measured altitude rate. Note: $\dot{H}$ is originally computed in the State Vector Update Routine, and is updated by P66 guidance.
$\dot{H}_D$	V	Desired altitude rate
Lag/TAUROD	P	Lag is the interval from the ROD PIPA read instant to the time the resulting thrust change is, in effect, achieved (i. e., the time an equivalent acceleration step is achieved as shown in Appendix A of MIT/DL Report E-2543.) The erasable padload Lag/TAUROD simplifies the LGC arithmetic.
LASTTPIP	V	The time of the preceding P66 PIPA readings.

NAME	TYPE	DEFINITION
MAXFORCE, MINFORCE	P	Thrust limits of the Throttle-Command Routine.
$\underline{\mathrm{OLDPIPA}}_P$	V	The PIPA readings of the current P66 cycle.
OMEGAQ	V	The current autopilot estimate of the attitude rate about the Q (or Y) body axis.
$\underline{\mathrm{PIPA}}_P$	V	Registers which collect the accelerometer counts. These are reset to zero when read by the State Vector Update Routine, but are not reset when read by P66.
$\underline{\mathrm{PIPATMP}}_P$	V	PIPA read temporaries for P66 velocity update and landing analog displays. Filled by the State Vector Update Routine and zeroed by it when it transfers the temporary state vector $\underline{r}'_P$, $\underline{v}'_P$ to the permanent registers $\underline{r}_P$, $\underline{v}_P$.
PRIORITY	V	Priority level assigned to each job scheduled in the AGC.
QHZ	P	A gain constant for horizontal nulling.
RIMUZ	F	The Z coordinate of the IMU in the LM body frame.
RODCOUNT	V	Count of the number of manipulations of the ROD controller since the preceding pass through the ROD equations or, on the first ROD pass, since the final pass through the P63 or P64 guidance equations.
RODSCALE	P	Altitude rate increment produced by one manipulation of the Rate-of-Descent Controller.
$\underline{\mathrm{RUPTREG}}_P$	V	The PIPA readings of the preceding P66 cycle.
TAUHZ	P	Inverse gain constant for horizontal nulling.
TAUROD	P	Inverse gain of ROD guidance. This is the time constant of the exponential nulling of the altitude rate error.
$\underline{\mathrm{TEM}}_P$	V	PIPA read temporaries for P66 thrust acceleration computation. Filled by the State Vector Update Routine and zeroed by P66 vertical channel after use.
THISTPIP	V	The time of the current P66 PIPA readings.
TOOFEW	P	One less than the minimum number of throttlings between P66 deletions.

NAME	TYPE	DEFINITION
TTPIP	V	t - PIPTIME: Time interval since the validity time of the current state vector $\underline{r}_P$, $\underline{v}_P$.
$\underline{u}_{FDP}$	V	Thrust direction input to FINDCDUW. Also stores the last valid thrust acceleration vector command from P63, P64, or P66.
$\underline{v}_{DP}$	V	Commanded velocity vector for horizontal guidance. Initialized at approximately lunar surface rate.
$\underline{v}66_P$	V	Velocity vector computed by P66 valid at P66 sample instant.
ξ	V	Angle between X-axis of vehicle and the local vertical.
2LATE466	P	Critical time interval between current time and PIPTIME of current servicer cycle beyond which one P66 pass is skipped (two ROD passes are skipped).

5.3.4.7 Throttle-Command Routine

The Throttle-Command Routine determines the thrust increment Δf_{th} required to achieve the thrust acceleration command s issued by the Guidance-and-Control Routine, and issues the corresponding command (ΔBIT, a bit count increment) to the Descent Engine Control Assembly.

The Throttle-Command Routine operates the engine in the throttleable region or at maximum thrust. The throttleable region is limited by manual throttle voltage at the lower end, and by mixture ratio variations and radically increasing erosion at the higher end. The upper limit of the throttleable region is defined by the switching criterion FHI, which is defined as that thrust above which the engine must not operate unless the throttle is moved to its maximum position, limited by a hard mechanical stop.

An information-flow diagram for the Throttle-Command Routine is presented as Figure 3.4.7-1. The basic characteristics of this routine as are follows:

An initial value of command thrust f is first computed, using the current estimated vehicle mass m and the command thrust acceleration s. When it is desired to place the throttle at its maximum setting, a command thrust augment δf_{sat} is required to insure that the throttle is kept and held against the mechanical stop.

The measured thrust $\tilde{f}$ corresponding to the present throttle setting is computed using the magnitude of the measured PIPA-output data $\Delta \underline{\tilde{v}}_P$ for the interval Δt between the last two PIPA-processing times. Because of computation delays and other factors, the change in throttle setting is not actually accomplished until close to the subsequent PIPA processing time. Therefore the term $\left| \Delta \underline{\tilde{v}}_P \right| m/\Delta t$ is not an accurate measurement of thrust at the current throttle setting unless the throttle setting was not changed the previous guidance cycle. Accordingly, if the previous throttle cycle was within three seconds of the current cycle, a correction term δf_p is added to improve the accuracy of the measurement. The subscript p is used to indicate that the correction term used on the current cycle through the Throttle-Command Routine was computed on the preceding cycle.

In order to determine whether to operate in the throttleable region or at maximum throttle, the Throttle-Command Routine compares the present thrust command f, and the previous thrust command f_p, against the switching criterion FHI, and a somewhat lower criterion FLO. If the throttle is in the throttleable region, these tests cause the throttle to remain in the throttleable region as long as the thrust command remains below FHI. If the throttle is at maximum, these tests cause the throttle to remain at maximum as long as the thrust command remains above FLO. The separation between FLO and FHI assures the throttle will never oscillate between the maximum setting and the throttleable region.

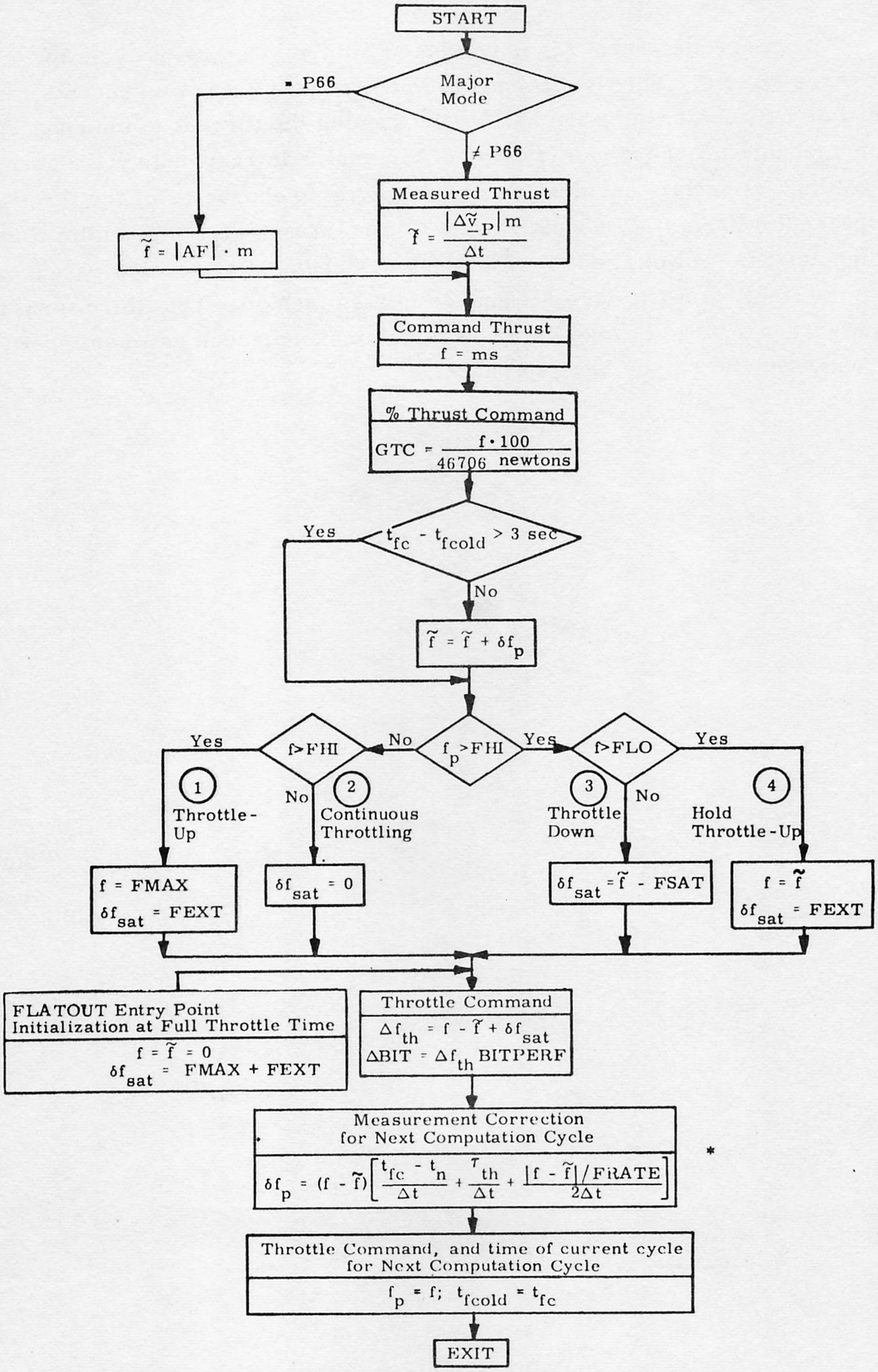


Fig. 3.4.7-1 Throttle-Command Routine

*Δt is the period of the guidance cycle of the guidance program calling the Throttle Command Routine. Thus in P66, Δt is half its value in other guidance programs.

The correction term δf_p is computed for the subsequent cycle to account for three effects. The first term within the square brackets accounts for the elapsed time between reading the PIPAs and issuing the throttle command. The second term accounts for the dynamic delay of the engine in responding to a change in throttle setting. The third term accounts for the ramp character of the throttle command proceeding from the Descent Engine Control Assembly, i.e., following the LGC computation, the voltage from this device changes at a constant rate from the initial to the final value, it does not change in step fashion. This third term is important only when a large change in thrust is commanded, such as when throttle control is recovered during the braking phase.

5.3.4.8 Landing Maneuver Display Computations

During the lunar landing maneuver, the LGC is required to provide digital displays on the DSKY and drive analog-type vehicle displays. The digital displays are updated at the normal PIPA-output processing times, i.e., every two seconds, with the exception of H and $\dot{H}$, which are updated once per second in P66. The analog displays are updated at approximately .25-second intervals between the normal PIPA processing times.

DSKY Display Parameters

The following digital display parameters are required for the lunar-landing programs P63, P64, and P66 as automatic or callable DSKY displays as specified in Section 4:

1.) H: Altitude
The estimated local altitude of the vehicle as derived from the difference between the magnitude of the current position vector ($\underline{r}_P$) and the landing-site radius magnitude ($\underline{r}_{LS}$).

2.) $\dot{H}$: Altitude Rate
The component of the vehicle's velocity that lies along the local-vertical direction ($\underline{u}_{hP}$).

3.) ABVEL: Inertial Velocity
The magnitude of the vehicle's inertial velocity.

4. ΔV_M: Measured or Accumulated ΔV
The total required velocity increment (ΔV) accumulated to the current time during the powered landing maneuver.

5. TFI: Time from DPS Ignition
The difference between the present time (t) and the DPS ignition time (t_{IG}), where t_{IG} is the time at which the trim phase is started.

6. Δh: Altitude Difference

The difference between the measurement of local altitude as derived from LR range data ($\tilde{h}$) and the altitude derived from the extrapolated state estimate (h') as modified by the a priori terrain model.

7. TTT: Time relative to achieving target conditions.

8. ΔT_{REDES}: Time remaining to redesignate the landing site.

9. v_{HF}: Forward component of the horizontal velocity of the vehicle with respect to the moon. Computed by same equations that compute v_{HF} for analog meters.

10. OUTOFPLN: Out-of-plane or cross-range distance between the initial LM descent orbital plane and the original landing site. This parameter, displayed to the nearest 0.1 n.mi., represents the cross-range distance that will be traversed during the powered landing maneuver. A positive sign signifies a landing trajectory out-of-plane change to the north, negative to the south. This computation is performed one time only, at the start of P63.

11. RANGE: Range to the Landing Site

The horizontal range from the LM to the current landing site.

12. $\theta e \ell$: LPD Angle

The depression angle of the current landing site referenced to the Landing Point Designator.

The computations required for the display quantities H, $\dot{H}$, ABVEL, ΔV_M, and Δh are shown in the State-Vector-Update Routine information-flow diagram. H and $\dot{H}$ are also computed in the Guidance and control routine, as are TTT, ΔT_{REDES}, OUTOFPLN, RANGE and $\theta e \ell$.

Analog Display Parameters

The Landing Analog Displays Routine, R10, drives four analog displays during landing, ascent, and abort maneuver phases. Computations are performed four times per second, but the results are displayed only if the crew places the MODE SEL switch in the PGNS position. The following quantities are displayed:

1.) h: Altitude
The estimated local altitude of the vehicle as derived from the difference between the magnitude of the current position vector ($|\underline{r}_P|$) and the magnitude of the landing site radius ($|\underline{r}_{LS}|$). The LGC drives a digital tape meter with altitude scaled at 2.345 feet/bit over a range of 0–60,000 feet.

2.) v_V: Altitude Rate
The component of the vehicle's velocity that lies along the local-vertical direction ($\underline{u}_{hP}$). The LGC drives a digital tape meter with this parameter scaled at 0.5 ft/sec per bit over a range of ±500 ft/sec.

3.) v_{HF}, v_{HL}: Forward and Lateral Velocity
Orthogonal components of the horizontal velocity of the vehicle with respect to the moon, which are essentially parallel and perpendicular to the X - Z plane of the vehicle (for small pitch and roll angle displacements). The coordinates and display requirements for these velocity components are defined in ICD LIS-540-10001. The scaling of these components is for a range of ±200 ft/sec.

The State Vector Update Routine provides the following quantities needed for the displays. These quantities are updated every 2 seconds, and are valid at PIPTIME, the time the PIPAs are read.

$\underline{v}_P$	-	Inertial velocity of LM (Stable Member Coordinates)
($\underline{G}$-$\underline{V}$BIAS)	-	Acceleration of Gravity minus PIPA bias
$\underline{P}IPATMP_P$	-	Contents of PIPAs at PIPTIME
$\underline{u}_{hP}$	-	Unit position vector at PIPTIME

H	-	Altitude
$\dot{H}$	-	Altitude Rate
DALTRATE	-	Rate of change of $\dot{H}$ due to change in direction of $\underline{u}_{hP}$
$\underline{V}$SURF	-	Lunar Surface Velocity

Also available are the present PIPA readings ($\underline{PIPA}_P$) and the Stable Member Outer Gimbal Angle (AOG).

R10 keeps the displays up to date between state vector updates by using PIPA data at intermediate times, t_i. In the following equations, the subscript i refers to the display-computation time.

For each R10 update, the vehicle's velocity with respect to the moon, $\underline{v}_{MP}$, must first be computed:

$$\underline{v}_{MP_i} = \underline{v}_P + (\underline{G}-\underline{V}BIAS)\,(t_i\text{-PIPTIME}) + \underline{P}IPA_P + \underline{P}IPATMP_P - \underline{V}SURF$$

From this can be calculated the vertical velocity, v_V:

$$v_{V_i} = \underline{v}_{MP_i} \cdot \underline{u}_{hP} + (\text{DALTRATE})\,(t_i\text{-PIPTIME})$$

The vehicle altitude, h, can then be computed:

$$h_i = H + \frac{v_{V_i} + \dot{H}}{2}\,(t_i\text{-PIPTIME})$$

Finally forward and lateral velocity components (v_{HF} and v_{HL}) are computed using the assumption that the cross range unit vector corresponds to the stable member Y axis; and therefore, the downrange unit vector is related to the unit position vector, $\underline{u}_{hP}$.

$$v_{HZ_i} = - v_{MPX_i}\, u_{hPZ} + v_{MPZ_i}\, u_{hPX}$$

$$v_{HY_i} = v_{MPY_i}$$

$$v_{HF_i} = v_{HZ_i} \cos AOG_i - v_{HY_i} \sin AOG_i$$

$$v_{HL_i} = v_{HZ_i} \sin AOG_i + v_{HY_i} \cos AOG_i$$

For ascent or aborts, the last two equations are replaced by:

$$v_{HL_i} = v_{MPY_i} + VSURF_Y \qquad \text{and} \qquad v_{HF_i} = 0$$

5.3.4.9 Landing Confirmation Routine and Post-Landing Sequences

When the lunar probe indicates contact with the lunar surface, the crew shuts down the DPS and waits for touchdown. When touchdown has been confirmed, the Landing Confirmation Routine (P68) is entered via the DSKY to terminate all of the landing maneuver routines. The Average-G Routine is allowed to cycle to the next PIPA processing time to establish the final landing condition state vector. The lunar surface flag is set, and the final LM position $\underline{r}_P$ is transformed to the Moon-Fixed Coordinate System as $\underline{r}_{LS}$, and stored for following prelaunch programs. While on the surface, the Planetary Inertial Orientation Subroutine is used to advance the landing site position vector $\underline{r}_{LS}$ to designated times for computations performed in Basic Reference Coordinates involving the LM state vector while on the lunar surface.

The vehicle attitude in lunar-fixed coordinates is determined as in the Lunar Surface Alignment Program P57 and stored. Before PGNCS power down, an IMU fine alignment is made and the vehicle attitude redefined and stored in place of the previous attitude. These vehicle attitude vectors (the body X- and Z-axis directions) are used in the following IMU alignment operations. An AGS initialization operation is normally performed prior to PGNCS power down. The LGC computed latitude, longitude and final site altitude are displayed on the DSKY.

(Pages 5.3-122, 5.3-123 and 5.3-124 intentionally omitted.)

5.3.5 POWERED ASCENT GUIDANCE

5.3.5.1 Guidance Objective

The objective of the Powered Ascent Program (P-12) is to control the LM ascent maneuver to achieve a specified velocity vector at a desired altitude relative to the launch site and desired distance out of the CSM orbital plane. In order to control the ascent maneuver to these three velocity and two position injection constraints, an explicit guidance concept is used which employs a linear control law. The engine configuration allows control of the direction of thrust only; thus explicit control of only radial (R) and cross range (Y) position and velocity is possible. The desired velocity in the down range (Z) direction is achieved by terminating the thrust at the proper time. The development of the ascent guidance equations is such that the best performance is achieved when the required velocity change is in the down range (Z) direction. In the lunar launch or abort maneuvers, controlled by this program, all injection position and velocity parameters are therefore controlled with the exception of the position along the velocity vector, referred to as the down range position.

The Powered Ascent Guidance Program can control LM ascent maneuvers initiated from non-coplanar launch conditions to be coplanar with the CSM orbital plane at injection, or parallel to the CSM orbital plane at a specified out-of-plane or cross range distance if the astronaut does not wish to remove all of the launch out-of-plane distance during the ascent maneuver. The program is also designed to achieve ascent injection under some off nominal APS thrust conditions (e.g. a single APS helium tank failure), and to allow for RCS injection in cases of premature APS shut down.

The Powered Ascent Guidance Program consists of three major phases.

1) Pre-Ignition Phase

2) Vertical Rise Phase

3) Ascent Guidance Phase

The computational procedures (Section 5.3.5.9) presented in this section are also used during abort maneuvers initiated from the powered lunar landing maneuver. These abort maneuvers can use either the DPS, APS, or a staged combination of the two to achieve the desired abort injection conditions. The abort programs and targeting used to control abort maneuvers using the Powered Ascent Guidance are presented in Section 5.4.3.

5.3.5.2 Ascent Guidance Coordinate Systems

The Powered Ascent Guidance Program uses two special coordinate systems in addition to the coordinate systems defined in Section 5.1.4.

One is an instantaneous local vertical coordinate system which is used in the velocity control portion of the ascent guidance computations. This coordinate system was selected because the radial rate and horizontal component of velocity for a vehicle in a circular orbit remain nearly fixed in the absence of thrust acceleration.

The other is an inertial moon centered pseudo coordinate system in which one axis is an arc length. This system is referred to as the Target Coordinate System.

Both coordinate systems are illustrated in Fig. 3.5-1 and are defined as follows.

Target Coordinate System

The target coordinate system is referenced to the CSM orbital plane and initial LM position by:

$$\underline{Q} = \text{UNIT}\,(\underline{v}_C \times \underline{r}_C)$$

$$\underline{S} = \text{UNIT}\,(\underline{r}_0 \times \underline{Q}) \qquad (3.5.1)$$

$$\underline{P} = \underline{Q} \times \underline{S}$$

where $\underline{r}_C$ and $\underline{v}_C$ are the CSM position and velocity vectors, and $\underline{r}_0$ is the initial LM position vector at t_{IG}. The vehicle Y position is measured from the PS plane as shown in Fig. 3. 5-1 as an arc length.

Local Vertical Coordinate System

The local vertical coordinate system of Fig. 3. 5-1 is defined by:

$$\underline{u}_R = \text{UNIT } (\underline{r})$$

$$\underline{u}_Z = \text{UNIT } (\underline{u}_R \times \underline{Q}) \qquad (3.5.2)$$

$$\underline{u}_Y = \underline{u}_Z \times \underline{u}_R$$

where $\underline{r}$ is the current LM position vector. The vector $\underline{r}_{CO}$ (Fig. 3. 5-1) is the LM position vector at engine cut-off.

5.3.5.3 Input - Output

The following input parameters are required by the Ascent Guidance Program in addition to the CM and LM state vectors normally stored and updated in the LGC.

Input Parameters

1. t_{IG} Ignition time

2. R_D Desired injection radius (Nominally 60, 000 ft. larger than tho landing site radius)

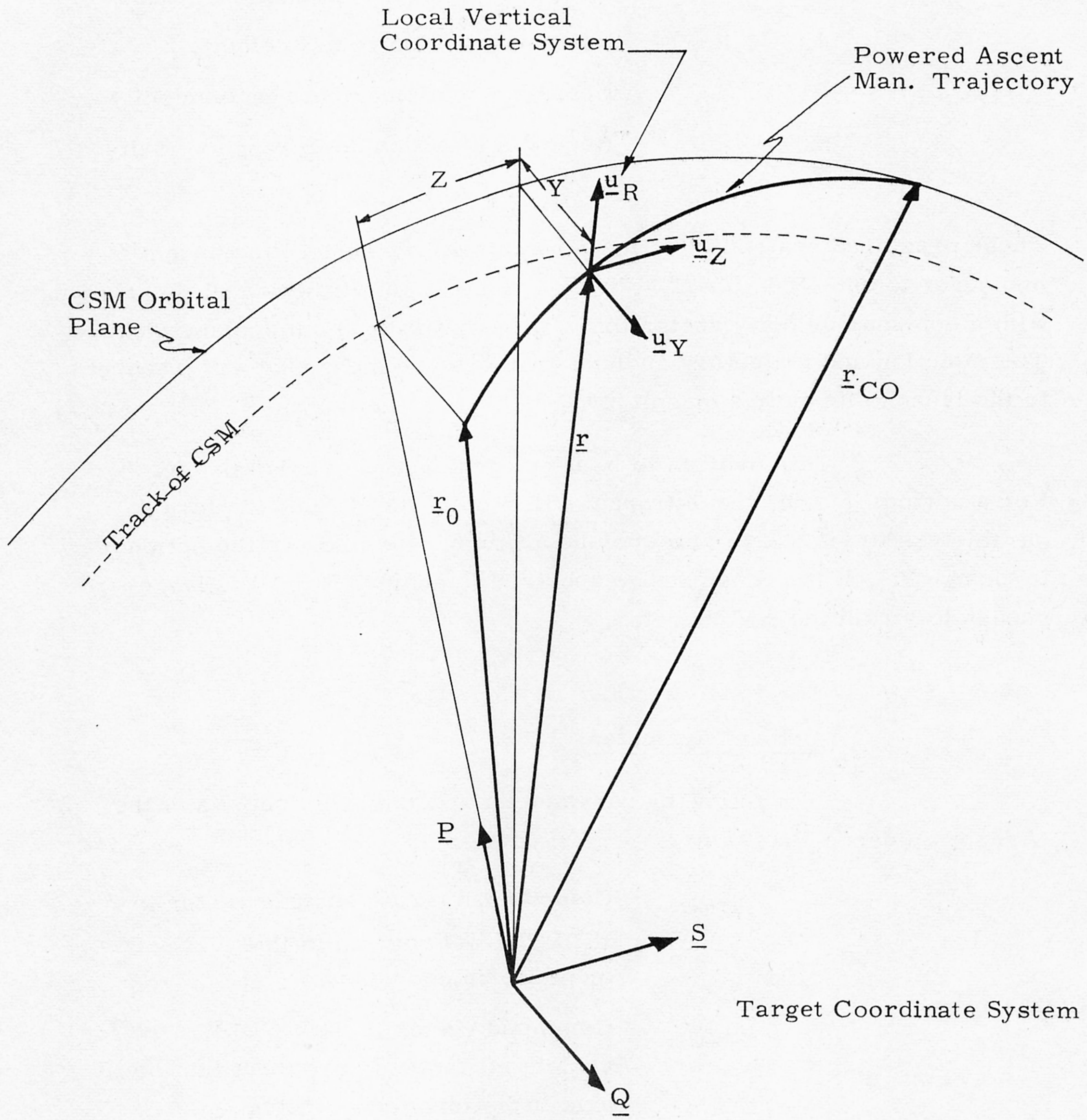


Fig. 3.5-1 Powered Ascent Guidance Coordinate Systems

3. Y_D Desired injection out-of-plane distance, measured from the CSM orbital plane

4. $\dot{R}_D$ Desired injection radial velocity

5. $\dot{Y}_D$ Desired injection cross range velocity

6. $\dot{Z}_D$ Required injection down range velocity

These prestored desired injection parameters will control the ascent maneuver to cut-off in the CSM orbital plane at an altitude of 60,000 ft. with a coplanar velocity vector that will result in a 30 nm apolune altitude. The injection and trajectory apolune altitudes are measured with respect to the launch site radius magnitude.

This orbit is to be used for a "quick, early takeoff." For a normal launch, the astronaut will modify the $\dot{R}_D$ and $\dot{Z}_D$ target parameters to yield a 45 nmi apolune altitude. He also has the option to change Y_D, if the cross-range required to achieve $Y_D = 0$ is large enough to strain the ΔV budget.

Output Parameters

The following parameters are the basic outputs of the Ascent Guidance Program:

1. $\underline{u}_{FDP}$ Commanded thrust attitude vector to FINDCDUW Program in IMU, or platform, coordinates

2. $\underline{u}_{WDP}$ Command vector to FINDCDUW used for vehicle attitude control about the thrust axis in platform coordinates

3. Engine-off Signal

4. The following display parameters (See Section 4, P-12):

Pre-Ignition Phase

a)	t_{IG}	Input ignition time initially displayed for verification
b)	CR	Cross range distance from the launch site to the CSM orbital plane that will be taken out during the ascent maneuver
c)	$\dot{Z}_D$	Down range injection velocity, as initially targeted
d)	$\dot{R}_D$	Injection radial rate, as initially targeted
e)	FDAI Yaw Angle	The yaw angle that should be present on the FDAI at the end of the vertical rise phase prior to pitch over
f)	FDAI Pitch Angle	The FDAI pitch angle that should be present after pitch over has been completed in the early portion of the ascent guidance phase
g)	TFI	Time from Ignition

Vertical Rise and Ascent Phase

a)	$(v_{GB})_X$	X-component of velocity to gain, in body coordinates.
b)	$\dot{h}$	Altitude Rate
c)	h	Altitude or radial position measured from the landing site radius

Callable Displays

Set 1	a)	t_{go}	Time from cut-off
	b)	$(v_{GB})_Y$	Y-component of velocity to gain, in body coordinates.
	c)	v	Inertial speed
Set 2	d)	$\underline{v}_{GB}$	Velocity to gain, in body coordinates

5.3.5.4 Powered Ascent Program P12 Pre-Ignition Phase

The pre-ignition computation sequence is shown in Fig. 3.5-2. With reference to this figure, after the ignition time, t_{IG}, has been verified by the astronaut, the following is a definition of the parameters initialized (numeric values are found in Section 5.8).

$a_T(APS)$	=	Initial APS thrust acceleration
τ (APS)	=	Initial APS mass to mass flow rate ratio
$\frac{1}{\Delta V}$	=	Initial velocity parameter for the initialization of $\frac{1}{\Delta V_1}$, $\frac{1}{\Delta V_2}$, and $\frac{1}{\Delta V_3}$ in the thrust magnitude filter (Section 5.3.5.9)
$\Delta t_{Tail\text{-}off}(APS)$	=	A negative time increment used to correct t_{go} for the APS tail-off
V_e (APS)	=	APS exhaust velocity

The flag, FLPI, is set to 1 to provide for the use of the Ascent Guidance Equations as a subroutine, predicting the commands at tipover.

Three of the five desired injection target parameters (R_D, Y_D, $\dot{Y}_D$) are then initialized as shown in Fig. 3.5-2. After the target vector $\underline{Q}$ is computed and the time-to-go, t_{go}, is set to a nominal value of 370 sec., the LM position vector is advanced to the ignition time, t_{IG}, by the Planetary Inertial Orientation Subroutine of Section 5.5.2. The velocity $\underline{v}(t_{IG})$ is updated as a function of the landing or launch position determined by the Planetary Inertial Orientation Subroutine as noted in Fig. 3.5-2 where ω_M represents the lunar inertial rotation rate and the unit vector $\underline{u}_M$ represents the lunar axis of rotation. The remaining two targets, $\dot{R}_D$ and $\dot{Z}_D$, are obtained from fixed storage.

During the initialization, t_{XO}, the X-axis override time is set to 0. This operation is a remnant from an earlier version of Luminary in which it was possible to enter the Ascent Guidance Equations with FLVP = 0.

The initially-displayed cross-range distance, CR, to be achieved by the ascent maneuver is based on injection coplanar with the CSM ($Y_D = \dot{Y}_D = 0$). If the astronaut does not desire to have the entire launch out-of-plane distance taken out during the ascent maneuver, he can specify how much out-of-plane distance should be removed during the powered ascent by overwriting the initial CR DSKY display. Injection conditions will then be parallel, but non-coplanar with respect to the CSM orbit plane.

The radial and down-range velocity targets are those pre-stored for quick liftoff, and in a normal launch would be modified by the astronaut.

Equation (3. 5. 3) deleted by PCR #270.

With reference to Fig. 3. 5-2, the LM state vector is next extrapolated through a nominal vertical rise phase so that initial ascent guidance phase command acceleration vectors can be calculated for the pre-ignition phase FDAI displays. These ascent phase acceleration vector computations are described in Section 5. 3. 5. 9 and in Fig. 3. 5-3. The FDAI yaw and pitch angle display parameters determined by Eqs. 3. 5. 21 and 3. 5. 22 respectively represent the FDAI yaw angle that will be achieved at the end of the vertical rise phase assuming that attitude control about the thrust axis is completed in this phase, and the FDAI pitch angle that will result early in the ascent guidance phase after pitch-over transients have damped out. The time from ignition, TFI, is the final display computation made in Fig. 3. 5-2.

The final operations during the pre-ignition phase are to extrapolate the LM position vector to a condition 30 seconds before ignition, t_{IG} - 30, and establish the LM state vector parameters $\underline{r}(t_{IG} - 30)$ and $\underline{v}(t_{IG} - 30)$. The vertical rise phase control flag, FLVP, is set, and the Average-G Routine of Sec. 5. 3. 2 is started at t_{IG} - 30. The Average G Routine is cycled at 2 second intervals initiating the computation sequence illustrated in Fig. 3. 5-3 at point B. The APS ignition is set for t_{IG}.

Nominal Powered Ascent P-12

Display t_{IG} previously loaded
prior to earth launch or updated
for
Astronaut verification or modification of t_{IG}

$FLPI = 1$
$FLVP = 1$
$FL4JET = 1$
$FLRCS = 0$

Initialize Thrust Magnitude Filter and DAP Parameters

a_T (APS)
τ (APS)
$1/\Delta V_1$, $1/\Delta V_2$, $1/\Delta V_3$
$\Delta t_{\text{Tail-off}}$ (APS)
V_e (APS)

Initialize Target Parameters

$R_D = r_{LS} + 60{,}000$ ft.
$Y_D = 0$
$\dot{Y}_D = 0$
$t_{XO} = 0$

Define Target Coordinate Vector $\underline{Q}$

$\underline{Q} = \text{UNIT}\,(\underline{v}_C \times \underline{r}_C)$

Set t_{go} to nominal value of 370 sec.

Advance LM Position Vector, $\underline{r}$, to t_{IG} obtaining $\underline{r}\,(t_{IG})$ and transform (0, 0, 1) to $\underline{u}_M$ by the Planetary Inertial Orientation Subroutine (Sec. 5.5.2)

$\underline{v}\,(t_{IG}) = \omega_M\,\underline{u}_M \times \underline{r}\,(t_{IG})$

1

Fig. 3.5-2 Powered Ascent Guidance P-12 Pre-Ignition Mode
(Page 1 of 2)

$\dot{R}_D = 19.5$ fps

$\dot{Z}_D = 5509.5$ fps

Compute Cross Range (CR) Display Parameter

$\underline{u}_r = \text{UNIT}\left[\underline{r}(t_{IG})\right]$

$Y = R_D(\underline{u}_r \cdot \underline{Q})$

$CR = Y_D - Y$

↓

Display CR, $\dot{R}_D$ and $\dot{Z}_D$ for astronaut approval or modification

↓

$Y_D = Y + CR$

↓

Extrapolate LM State Vector through Vertical Rise Phase

$\underline{r} = \underline{r}(t_{IG})$

$\underline{v} = \underline{v}(t_{IG}) + 49\ \underline{u}_r$ fps

→ Go to Point (C) of Fig. 3.5-3

(A) →

Compute and Display:

FDAI Pitch Angle

FDAI Yaw Angle

$TFI = t - t_{IG}$

↓

FLPI = 0

↓

Initialize FINDCDUW

↓

Extrapolate LM Position Vector to $t_{IG} - 30$ obtaining $\underline{r}(t_{IG} - 30)$ and transform (0, 0, 1) to $\underline{u}_M$ by the Planetary Inertial Orientation Subroutine (Sec. 5.5.2)

↓

$\underline{v}(t_{IG} - 30) = \omega_M \underline{u}_M \times \underline{r}(t_{IG} - 30)$

↓

Set APS Ignition at t_{IG}, Start Average G Routine at $t_{IG} - 30$,

↓

Go to Point (B) of Fig. 3.5-3

Fig. 3.5-2 Powered Ascent Guidance P-12 Pre-Ignition Mode (Page 2 of 2)

5.3.5.5 Powered Ascent Program P12 Vertical Rise Phase Control

The computation procedures during the vertical rise phase are described in Section 5.3.5.9. The proper path through these procedures is obtained by setting control flag FLVP = 1. Most of the procedures are explained in Section 5.3.5.9; the remainder is described below.

As shown on page 4 of Fig. 3.5-3 the nominal attitude X-axis override DAP mode is inhibited until 12 seconds after the vertical rise phase has been completed.

The vehicle attitude is controlled by FINDCDUW (Section 5.3.7) about the thrust or vehicle X axis such that the vehicle X - Z plane contains $\underline{u}_{WDP}$, the desired window pointing direction. During the vertical rise phase, $\underline{u}_{WDP}$ is defined to be the horizontal component of the desired acceleration computed by the guidance equations, $\underline{a}_T$, and the commanded thrust direction $\underline{u}_{FDP}$ is along the local vertical vector. These two vectors, $\underline{u}_{FDP}$ and $\underline{u}_{WDP}$, and the yaw-control mode flag FLAUTOX are the required input parameters to the LM digital autopilot interface subroutine designated FINDCDUW. The altitude check ($h \geq 25,000$) shown in Fig. 3.5-3 page 4 is used in the abort programs P-70 and P-71 (Section 5.4.3) to add an additional altitude constraint on the vertical rise phase.

5.3.5.6 Powered Ascent Program P12 Ascent Guidance Phase Control

The Ascent Guidance Phase is the third and final phase of the Powered Ascent Program P-12 and is initiated after the vertical rise phase is terminated. The computation procedures are described in Section 5.3.5.9. The proper path through these procedures is obtained by setting control flag FLVP = 0.

The flag, ROTFLAG, is provided for abort maneuver control, and is zero during P-12 operation. The operation of the equations with ROTFLAG = 1 is discussed in Section 5.4.3.1.4.

During the Ascent Guidance Phase the vehicle thrust attitude is commanded to be along the desired acceleration vector $\underline{a}_T$. The vehicle attitude about the thrust axis is commanded such that the LM windows or +Z axis will be in the downward direction. The X-axis override DAP mode is not activated until 12 seconds (t_{XO}) after the termination of the vertical rise phase to allow for attitude stabilization during the initial pitch-over maneuver. The astronaut can then manually control the vehicle yaw attitude for the duration of the powered ascent maneuver if desired. If the X-axis override mode is not exercised, the existing attitude orientation will be maintained.

The principal outputs of the program are the two attitude command vectors $\underline{u}_{FDP}$ and $\underline{u}_{WDP}$ to the DAP interface subroutine FINDCDUW (Section 5.3.7).

5.3.5.7 RCS Ascent Injection

If the APS is prematurely shut down prior to ascent injection or if it is desired to trim-up the ascent injection maneuver after nominal APS termination, the Ascent Guidance Program can be used to aid in the manual control of the RCS to achieve the desired velocity injection conditions. The computation sequence used in this type of operation is described in Section 5.3.5.9 and is illustrated in Fig. 3.5-3 page 1 with control flag FLRCS = 1. At APS termination the ΔV Monitor Routine places the vehicle in the attitude hold mode. The basic computation shown in Fig. 3.5-3 for the RCS mode of operation is the determination of the velocity-to-be-gained vector, $\underline{v}_G$. This velocity vector is transformed from IMU platform or stable member coordinates to vehicle body, or navigation base, coordinates (Section 5.6.3.2) and displayed so that the astronaut can manually complete the ascent maneuver. The RCS maneuver is done with 4 RCS jets, automatically selected by the setting of FL4JET at the beginning of P12 and P70/P71.

5.3.5.8 Abort Guidance from Lunar Landing Maneuvers

The DPS Abort Program P-70 and APS Abort Program P-71 control abort programs initiated during the powered lunar landing maneuver. The basic computation sequence used by both P-70 and P-71 is the Powered Ascent Guidance Computations of Section 5.3.5.9. The initialization of programs P-70 and P-71 is described in Section 5.4.3 and Fig. 4.3-1.

5.3.5.9 Ascent Guidance Computations Used by P12, P70 and P71

The guidance computation procedures presented in this section and Fig. 3.5-3 determine the desired thrust direction and time of thrust termination for both the vertical rise and ascent guidance phases of Program P12 (Powered Ascent) and Programs P70 and P71 (DPS and APS aborts).

The figure describing the computational flow of these procedures is Fig. 3.5-3 (pp. 1-4). The following set of switches establish the proper path thru these procedures for the various users:

Switch	Value	Meaning
FLRCS	1	RCS Injection Mode
	0	Not the RCS Injection Mode
FLENG1	1	Engine has been commanded on
	0	Engine has not been commanded on
FLENG2	1	An engine off call has been made
	0	An engine off call has not been made
FL4JET	1	4 RCS jets used in translation
	0	2 RCS jets used in translation
P7071FLG	1	Major mode is P70 or P71
	0	Major mode is P12
ROTFLAG	1	Rotation control is imposed
	0	FINDCDUW commands shortest rotation

FLVP	{	1	Vertical rise phase
		0	Not the vertical rise phase
FLPC	{	1	Injection position control not desired
		0	Injection position control desired
FLPI	{	1	P-12 pre-ignition phase
		0	Not the P-12 pre-ignition phase
FLDOC	{	0	LM is not docked to CSM

In addition to the above control flags, the control times t_2 and t_3 of Fig. 3.5-3 are used to regulate the guidance functions as follows:

t_2 = 2 sec: Represents the time interval before engine cut-off during which all guidance parameters are held at their last computed values.

t_3 = 10 sec: The time interval before engine cut-off during which the guidance parameters B and D that control the injection position in the R and Y directions are set to zero. During this 10 second interval all position control is terminated and the guidance system attempts to achieve the desired injection velocity conditions until t_2.

With reference to Fig. 3.5-3, the computation sequence is initiated at point (B) by the Average-G Routine every two second computation cycle. The following parameter computations are made in the sequence illustrated in Fig. 3.5-3 for the APS lunar ascent maneuver.

Thrust Filter Bypass

The existence of an engine on command is first checked and the velocity increment ΔV measured by the PIPA's over the last computation cycle is compared with a prestored minimum value ΔV_{min}. If the measured ΔV is less than ΔV_{min}, the X-component of velocity to gain, altitude rate and altitude are displayed and commanded attitude rates are zeroed. This check is in addition to that of the ΔV Thrust Monitor Subroutine of Section 5.3.3.6. The value of ΔV_{min} used in Luminary is 3.56 m/sec.

Thrust Filter Computations

Assuming a successful engine thrust test, the thrust filter computation is processed next. The vehicle velocity increments, ΔV, measured by the PGNCS IMU integrating accelerometers (PIPA's) include noise not only from the main engine, but also from the RCS jets used for attitude control. In order to estimate an accurate thrust acceleration performance (a_T and τ), several successive PIPA ΔV readings are averaged. Under the assumption of a constant mass flow engine, $1/a_T$ is a linear function of time. The ratio of mass to mass flow rate (τ) is then computed from four successive PIPA readings ΔV_0, ΔV_1, ΔV_2 and ΔV_3 as follows:

$$\Sigma = \frac{1}{\Delta V_0} + \frac{1}{\Delta V_1} + \frac{1}{\Delta V_2} + \frac{1}{\Delta V_3} \tag{3.5.4}$$

$$\tau' = \frac{\Sigma}{4} V_e \ \Delta t - 2 \ \Delta t \tag{3.5.5}$$

where Δt is the computation interval. In Eq. (3.5.5) the first term represents the estimate of τ' at the center of the series of time increments for which ΔV was measured, and the second term updates the estimate to current time. Additional smoothing is included by averaging this new value of τ' with the previous estimate of τ, updated by Δt.

$$\tau = \frac{1}{2} \left(\tau' + \tau - \Delta t \right) \tag{3.5.6}$$

The exhaust velocity V_e is prestored in the LGC. In the pre-ignition computation for lunar launch the three velocity increments $\frac{1}{\Delta V_1}$, $\frac{1}{\Delta V_2}$, and $\frac{1}{\Delta V_3}$ are prestored initialization constants. When a new $\frac{1}{\Delta V}$ is computed it is stored in the $1/\Delta V_3$ register, and the older values are shifted down. The vehicle thrust acceleration magnitude is then computed by

$$a_T = \frac{V_e}{\tau} \tag{3.5.7}$$

With reference to page 1 of Fig. 3.5-3, the following parameters are computed after the thrust filter computation.

Local Vertical Coordinate System

$$\underline{u}_R = \text{UNIT}(\underline{r})$$
$$\underline{u}_Z = \text{UNIT}(\underline{u}_R \times \underline{Q}) \qquad (3.5.8)$$
$$\underline{u}_Y = \underline{u}_Z \times \underline{u}_R$$

Local Velocity Components:

$$\dot{R} = \underline{v} \cdot \underline{u}_R$$
$$\dot{Y} = \underline{v} \cdot \underline{u}_Y \qquad (3.5.9)$$
$$\dot{Z} = \underline{v} \cdot \underline{u}_Z$$

Local Position Computation

The radial and cross range positions (Fig. 3.5-1) are computed by

$$R = r \qquad (3.5.10)$$
$$Y = R_D \sin^{-1}(\underline{u}_R \cdot \underline{Q})$$
$$\cong R_D(\underline{u}_R \cdot \underline{Q})$$

Effective Gravity Computation

$$g_{eff} = \frac{H^2}{r^3} - g_n \qquad (3.5.11)$$

where $H^2 = [\,|\underline{r} \times \underline{v}|\,]^2$ and g_n is the magnitude of gravity as computed by the Average-G Routine of Section 5.3.2.

Velocity to be Gained Computation

The velocity-to-be-gained vector $\underline{v}_G$ is next computed as shown in Fig. 3.5-3, page 1, and then transformed from the stable member coordinates in which the local vertical system is defined to the vehicle body or navigation base coordinate system by the transformation matrix [SMNB] as defined in Section 5.6.3.2. This velocity-to-be-gained vector referenced to navigation base coordinates is used for display parameters in the RCS injection mode.

Time-to-Go Estimation

As shown in Fig. 3.5-3, page 2, the time-to-go, t_{go}, is determined from a truncated series expansion of the exponential expression for t_{go} (v_G):

$$t_{go} = \tau \left(1 - e^{-\frac{v_G}{V_e}}\right) \qquad (3.5.12)$$

The series expansion used to approximate the above expression is truncated at the second order term.

$$t_{go} = \tau \frac{v_G}{V_e} \left(1 - \frac{1}{2} \frac{v_G}{V_e}\right) + \Delta t_{Tail\text{-}off} \qquad (3.5.13)$$

where

τ	=	$\dfrac{\text{mass}}{\text{mass flow rate}}$
V_e	=	Exhaust velocity
$\Delta t_{Tail\text{-}off}$	=	A negative time increment used to correct t_{go} for the thrust tail-off. For P12 the value used is $\Delta t_{Tail\text{-}off}$ (APS).

When the guidance is operating in the RCS mode, the time-to-go computation is simplified to

$$t_{go} = V_G / A_T \text{ (RCS)}$$

where A_T (RCS) is an assumed, constant, acceleration corresponding to 4 RCS jets in a dry LM. In situations where only 2 jets are used this leads to an underestimate of t_{go}. However, t_{go} is only used in the estimation of the gravity effect, and hence, is only a second order factor. The use of a 2-jet acceleration could lead to overflow in the computation of t_{go}.

Engine-off Test

On page 2 of Fig. 3.5-3 the engine-off command is set for $t + t_{go} - \Delta t_c$ when t_{go} first falls below 4 sec, where Δt_c is the computation time delay between the reading of PIPA's and the sending of the commands.

Guidance Parameter Computations

The basic guidance concept used for the powered ascent is represented by the equations:

$$\begin{aligned} a_{TR} &= a_T \left[A + B(t - t_o) \right] - g_{eff} \\ a_{TY} &= a_T \left[C + D(t - t_o) \right] \end{aligned} \qquad (3.5.14a)$$

where a_{TR} and a_{TY} are the required accelerations in the radial and cross range directions respectively, a_T is the filtered thrust acceleration, and t_o is the time at which the guidance coefficients A, B, C and D are computed.

There is a significant delay between the establishment of the state vector on which the commands are based and the sending of the commands by the autopilot. In addition, since the commands will be fixed for a 2-second period, while the guidance equations assume components varying linearly with time, the value of t should be selected in the middle of the 2-second command period, so that the actual commands are equal to the average of the assumed linear commands.

These delay times are all lumped together in the increment $(t - t_o)$, which is chosen to provide good guidance performance. In Luminary the present value of $t - t_o$ is fixed at 1 sec.

For satisfactory scaling within the LGC, the equations are reorganized into the form

$$\begin{aligned} a_{TR} &= \frac{1}{\tau} \left(A + [t - t_o] B \right) - g_{eff} \\ a_{TY} &= \frac{1}{\tau} \left(C + [t - t_o] D \right) \end{aligned} \qquad (3.5.14b)$$

where:

$$B = [D_{21} (\dot{R}_D - \dot{R}) - (R_D - R - \dot{R} t_{go})] / t_{go} E \qquad (3.5.15)$$

$$D = [D_{21} (\dot{Y}_D - \dot{Y}) - (Y_D - Y - \dot{Y} t_{go})] / t_{go} E \qquad (3.5.16)$$

$$A = - D_{12} B - (\dot{R}_D - \dot{R}) / L \qquad (3.5.17)$$

$$C = - D_{12} D - (\dot{Y}_D - \dot{Y}) / L \qquad (3.5.18)$$

where the parameters L, D_{12}, D_{21}, and E are defined on page 2 of Fig. 3.5-3 as previously mentioned. When the time-to-go, t_{go}, falls below 10 seconds (t_3) the rate coefficients B and D are set to zero, and the injection position control is terminated. When t_{go} falls below 2 seconds (t_2) no further guidance coefficient computations are made and the previous values are held for the remainder of the maneuver.

In order to make the response of PGNCS similar to that of the back up system, AGS, the pitch rate parameter, B, is limited by

$$-0.1 \text{ fps}^3 \tau \leq B \leq 0$$

It should be noted that the restriction of B to non-positive values causes the injection to occur above the specified 60,000 ft. altitude.

<u>Command Thrust Acceleration Vector Computation</u>

The required acceleration components a_{TR} and a_{TY} are computed as shown on page 3 of Fig. 3.5-3 for the R and Y injection parameters respectively. These two required acceleration components are then combined as

$$\underline{a}_H = a_{TY} \underline{u}_Y + a_{TR} \underline{u}_R \qquad (3.5.19)$$

$$a_H = | \underline{a}_H | \qquad (3.5.20)$$

and compared with a_T, where a_T is available thrust acceleration. If a_H is greater than a_T, the commanded $\underline{a}_H$ is reduced by a factor a_T/a_H and the Z component of $\underline{a}_T$ is set to 0. If the required a_H is smaller than a_T, the remainder $(a_T^2 - a_H^2)^{1/2}$ is directed along $\underline{Z}$ and $\underline{a}_H$ is unchanged. The sense of the Z-component of the computed thrust acceleration vector, $\underline{a}_T$, is determined by the sign of the velocity-to-be-gained in this direction SGN $(\dot{Z}_D - \dot{Z})$. The commanded thrust acceleration vector, $\underline{u}_{FDP}$, is set equal to the computed vector, $\underline{a}_T$, subject to the modifications discussed in Section 5.3.5.5.

FDAI Angle Computations

The Flight Director Attitude Indicator (FDAI) yaw angle at the end of the vertical rise phase, and the pitch angle after pitch-over and the start of the ascent guidance phase are required for display and monitoring purposes in the pre-ignition phase of program P-12 (Section 5.3.5.4). These FDAI display parameters are computed as follows, based on the lunar launch alignment (Section 5.1.4.2) of the IMU:

$$\text{FDAI Yaw Angle} = \sin^{-1}\left[\frac{a_{TY}}{(a_{TP}^2 + a_{TY}^2)^{1/2}}\right] \qquad (3.5.21)$$

if $(a_{TP}^2 + a_{TY}^2)^{1/2} = 0$ then Yaw Angle = 0

$$\text{FDAI Pitch Angle} = -\cos^{-1}\left[\text{UNIT}(\underline{u}_{FDP}) \cdot \underline{u}_R\right] \qquad (3.5.22)$$

Nominal Entry from Average-G Routine
Every 2 sec. Computation Cycle

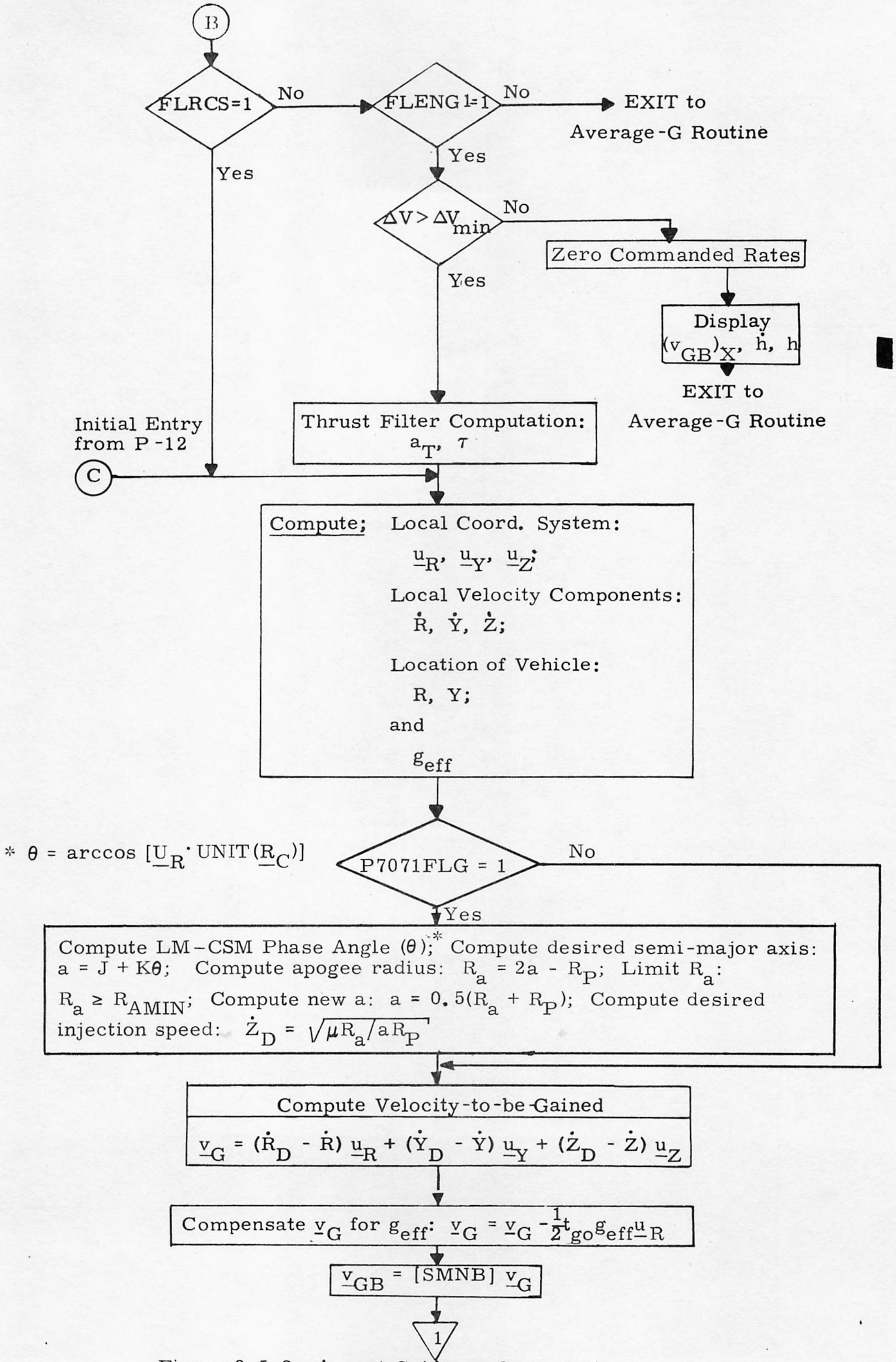


Figure 3.5-3 Ascent Guidance Computation Sequence (page 1 of 4)

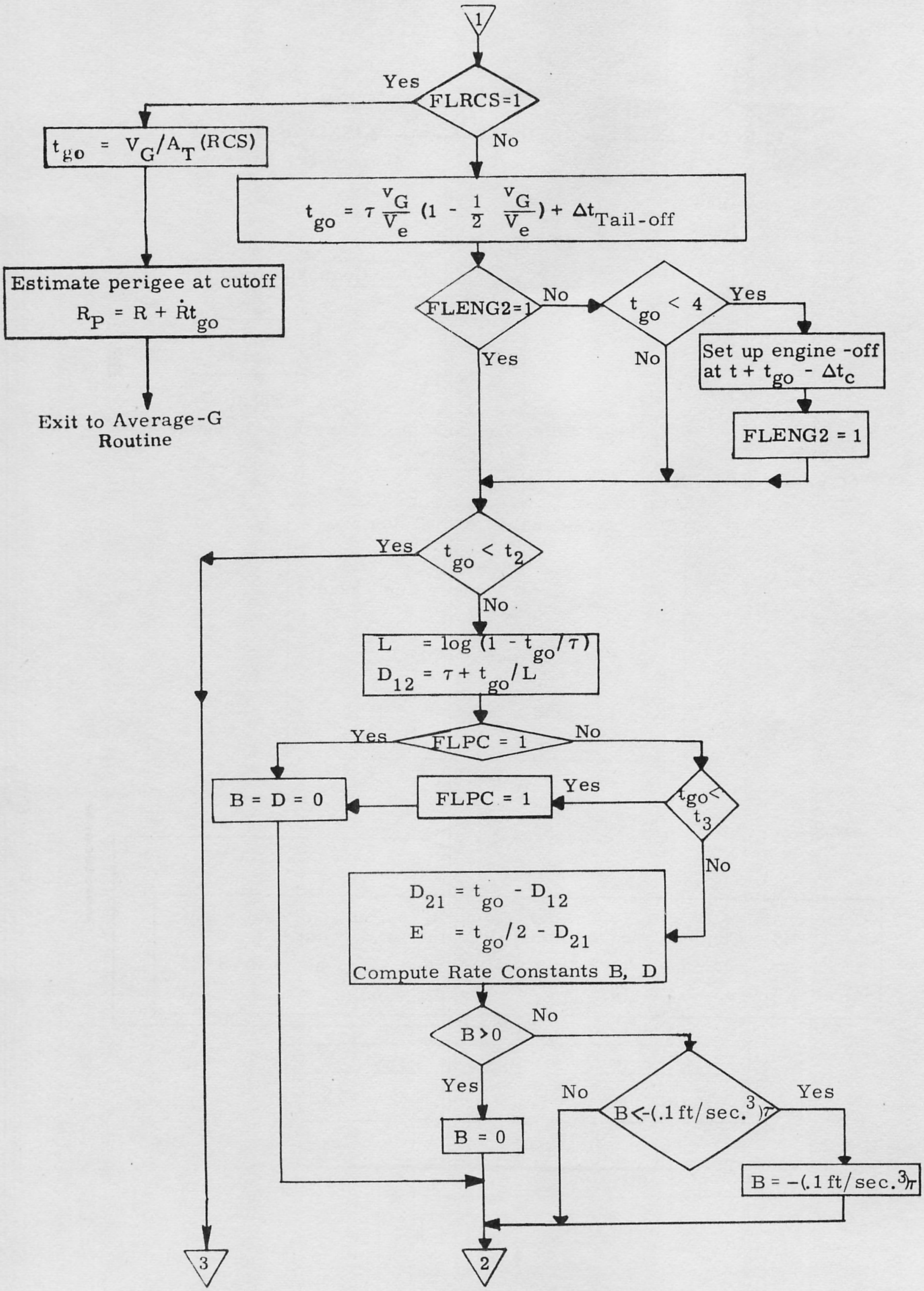


Figure 3.5-3 Ascent Guidance Computation Sequence (page 2 of 4)

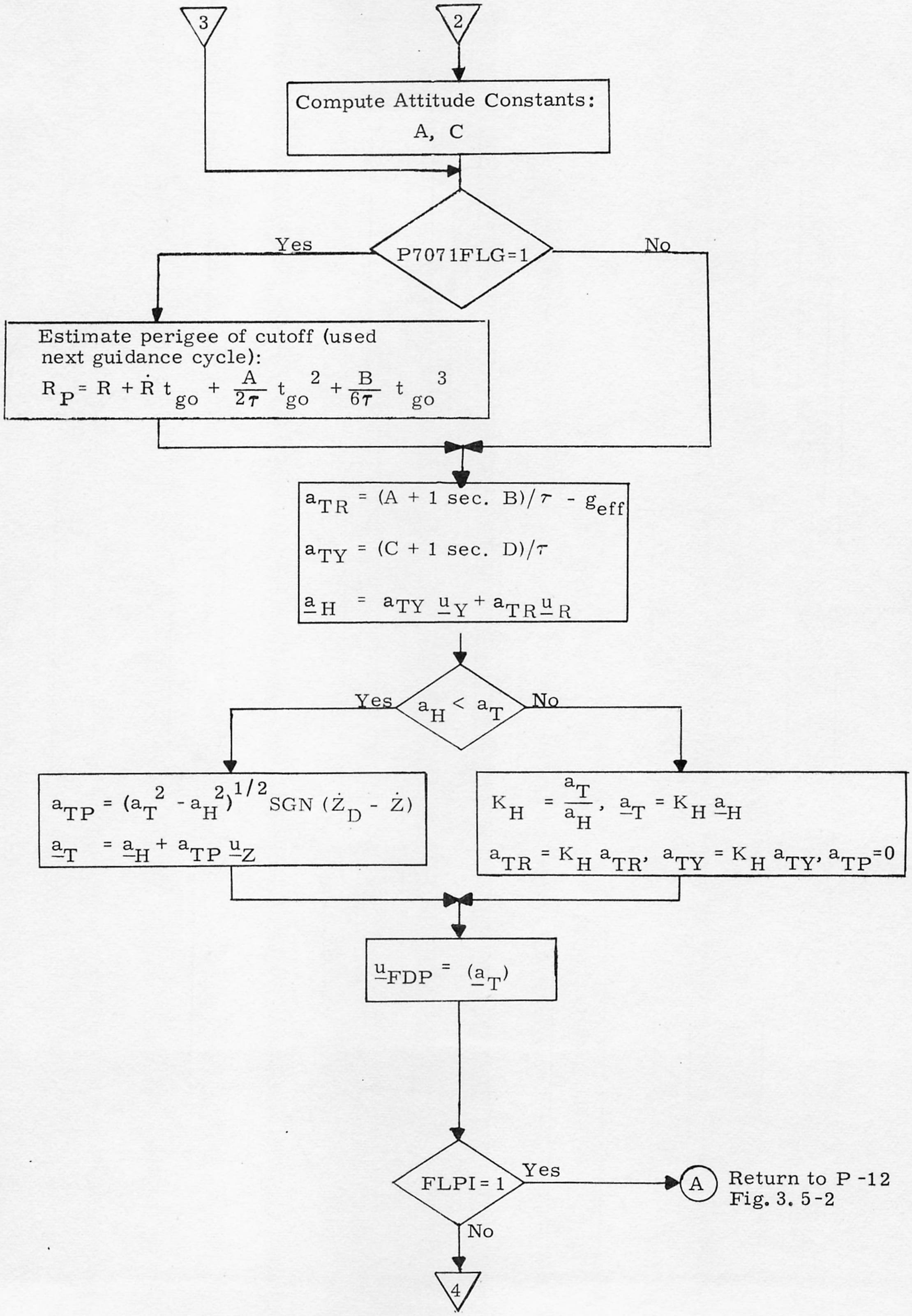


Figure 3. 5-3 Ascent Guidance Computation Sequence
(page 3 of 4)

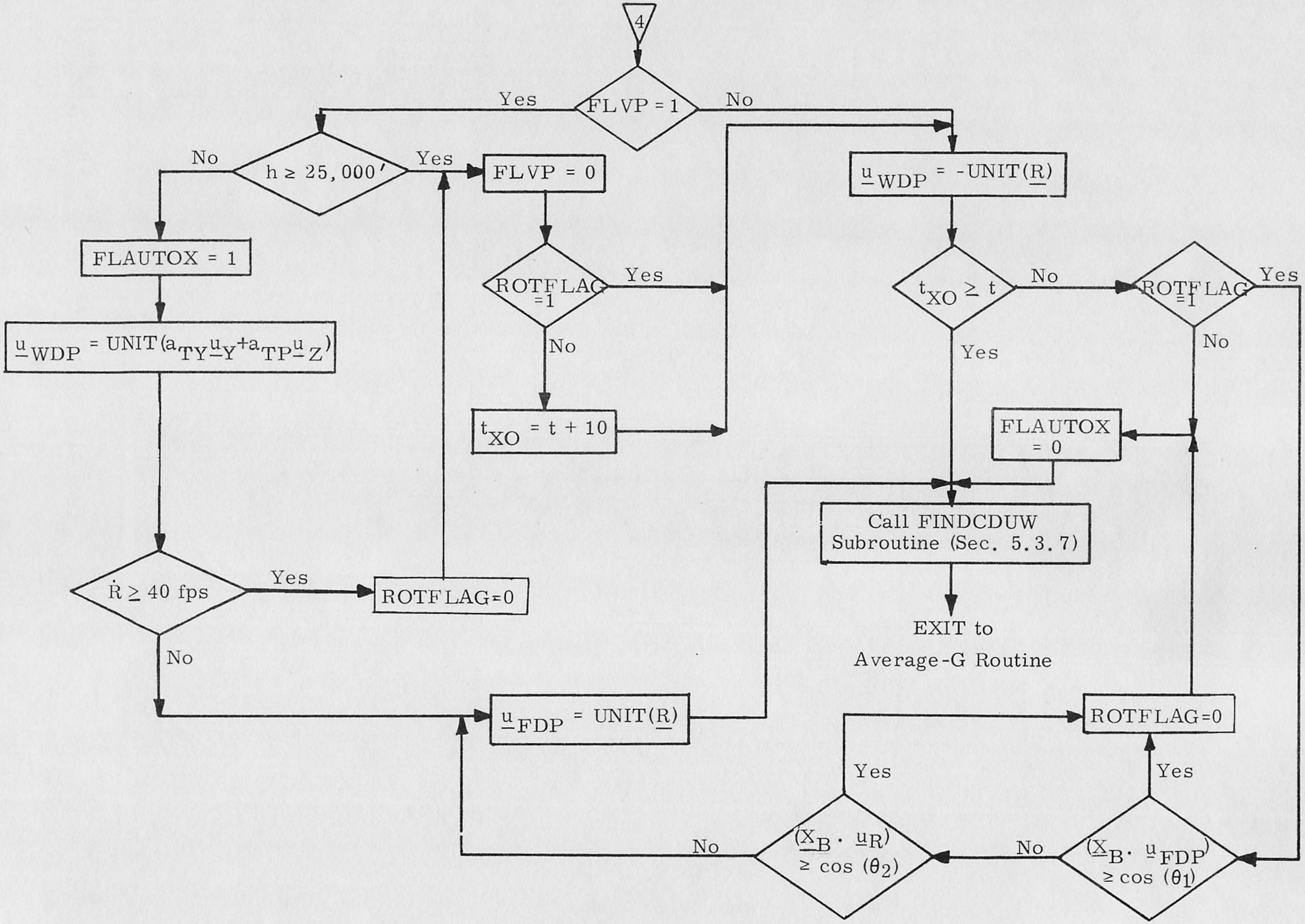


Figure 3.5-3 Ascent Guidance Computation Sequence
(Page 4 of 4)

5.3.6 THRUST MONITOR PROGRAM

The Thrust Monitor Program, P-47, is used during manual or non-PGNCS controlled maneuvers to monitor and display the velocity change applied to the vehicle. The program first suspends state vector updating by the radar data (resets the update and track flags), and advances the vehicle state vector to the current time by the Coasting Integration Routine of Section 5.2.2. This operation is continued until the state vector is advanced several seconds ahead of the current time as described in Section 5.3.8. The Average-G Routine of Section 5.3.2 is then initiated to allow thrusting to be started as soon as possible. The Average-G Routine is left on until the program is terminated after completion of the maneuver. The primary output of P-47 is the measured maneuver $\Delta \underline{V}$ in vehicle coordinates as described in Section 4.

The Thrust Monitor Program is normally used during the manually controlled terminal rendezvous phase. During this mission phase the Rendezvous Display Routine R-31 is normally called to display range, range rate and the line-of-sight angle θ. The operation of R-31 with the Average-G Routine of P-47 is described in Section 5.6.7.1.

5.3.7 FINDCDUW ROUTINE

5.3.7.1 Introduction

The purpose of this section is to describe the operation and implementation of the FINDCDUW Routine.

FINDCDUW is the interface routine between the various powered flight guidance programs and the digital autopilot. FINDCDUW receives from the guidance programs thrust and window pointing commands and provides the autopilot with gimbal angle increments, commanded attitude rates, and commanded attitude lag angles (which account for the angles by which the body will lag behind a ramp command in attitude angle - due to the finite angular accelerations available). FINDCDUW alines the estimated thrust vector with the unit thrust command vector, and, when in the automatic X axis control mode, alines the +Z half of the LM ZX plane with the window command vector.

FINDCDUW is designed to accomplish the following:

1. To produce the normal, small angle, maneuvers determined by the guidance inputs at essentially constant attitude rate, reaching the terminal attitude at the completion of the two second computation period.

2. To produce the gross maneuvers, which are required at the boundaries between guidance phases, or upon abort, with a continuous, rate-limited attitude profile extending over whatever time duration is required to complete the maneuver.

3. To avoid excessive middle gimbal angle in all maneuvers, never reaching the critical middle gimbal angle during a maneuver in which the terminal middle gimbal angle is less in magnitude than the critical value; to display an alarm code when inputs to FINDCDUW determine a terminal middle gimbal angle in excess of the critical value, and to limit the middle gimbal excursion at the critical value.

4. To permit manual X axis control when the flag FLAUTOX is 0, not interfering when this control is being exercised, not drifting about the X axis when manual control is permitted but not exercised.

5. To provide identical interfaces except with alternate rate limit values when the Command and Service Modules are docked.

6. To use the desired gimbal angles as maintained by the autopilot rather than the actual gimbal angles when transforming measured thrust to bodyaxes to avoid feeding fuel slosh oscillations into the powered flight portion of the system.

To accomplish item 6 above, two coordinate frames are defined. The first, called the current vehicle frame, is computed using the desired gimbal angles, CDUD's, maintained by the digital autopilot. The unit vectors of the current vehicle frame would be identically the body unit vectors if the desired gimbal angles were identically the actual gimbal angles. FINDCDUW provides a thrust direction filter which transforms the measured thrust acceleration vector to current vehicle coordinates and produces a filtered unit thrust vector in this coordinate frame $\underline{u}_{FV}$. The second frame, called the commanded vehicle frame, is computed using $\underline{u}_{FV}$, the thrust command $\underline{u}_{FDP}$, and the

window command $\underline{u}_{WDP}$. Since the commanded orientation is coupled to the actual orientation only through the low gain thrust direction filter, fuel slosh is not excited. Also, since the filter operates in the frame defined by the CDUD's, and the CDUD's are driven by the commands from FINDCDUW, the steady-state attitude error is zero.

The concept of FINDCDUW is as follows. The thrust and window pointing commands, and the location of the thrust vector relative to spacecraft axes, determine a required, or terminal, spacecraft attitude. The gimbal angles corresponding to this terminal attitude are computed and inputs to the autopilot are generated which drive each gimbal angle directly from its initial value to its terminal value*.

Provided the attitude is not initially in gimbal lock, and provided the input commands do not yield a terminal attitude in gimbal lock, it is impossible to pass through gimbal lock during the maneuver since the middle gimbal angle is confined to the range between its initial and terminal values.

A simplified flow diagram of the FINDCDUW Routine is shown in Fig. 3.7-1 with nomenclature defined in Section 5.3.7.2.

This routine is called every two seconds, which is the frequency at which the commanded vehicle attitude is computed.

---

*Except the outer gimbal angle profile may not be monotonic in the geometrically complex case of a large maneuver about multiple axes at substantial middle gimbal angle and with magnitude limiting of the X axis attitude angle change on at least one pass through FINDCDUW.

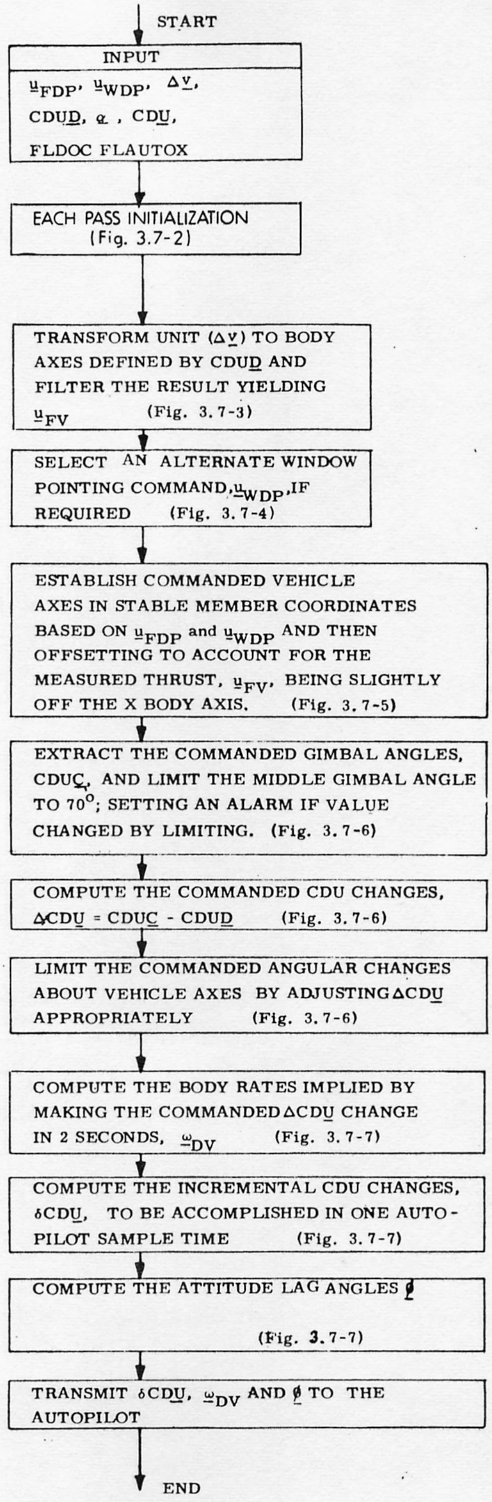


Figure 3.7-1 FINDCDUW Simplified Flow Diagram

5.3.7.2 Nomenclature

$\underline{u}_{FDP}$	Vector in the desired thrust direction in Platform Coordinate System. Need not be unitized.
$\underline{u}_{WDP}$	Vector in the desired window direction in Platform Coordinate System. Need not be unitized.
OGABIAS	Outer gimbal angle bias
$\Delta\underline{\tilde{v}}_P$	PIPA measured velocity change since the last sample period, platform coordinates.
CDU$\underline{U}$	Measured gimbal angles
CDU$\underline{D}$	The desired gimbal angles as maintained by the autopilot
$\underline{\alpha}$	The angular accelerations which can be achieved by the RCS, (2 Jet)
$\underline{u}_{FV}$	Filtered unit measured thrust direction vector
GAINFLTR	Gain of thrust direction filter
CDU$\underline{C}$	Commanded gimbal angles
$\underline{\omega}_{DV}$	Commanded body attitude rates sent to the autopilot
δCDU$\underline{U}$	Commanded incremental CDU changes sent to the autopilot to occur in one autopilot sample period

FLDOC	LM docked flag = 0 LM not docked ≠ 0 LM docked
FLAUTOX	X axis override flag = 0 override permitted ≠ 0 override not permitted
OVFIND	Computer overflow indicator
PGNCS MODE	LM console switch Yes = PGNCS control No = AGS control
AUTO MODE	LM console switch Yes = Auto Mode No = Attitude Hold Mode or Off Mode
$C_{PV}(CDU\underline{V})$	3 × 3 matrix which transforms a vector in platform coordinates to the vehicle coordinates defined by $CDU\underline{V}$ (Fig. 3.7-2)
$\underline{u}_{ZBP}$	Unit Z body axis, platform coordinates (supplied by service routine)
$\underline{u}_{XBP}$	Unit X body axis, platform coordinates (supplied by service routine)

5.3.7.3 Detailed Flow Diagrams

5.3.7.3.1 First Pass Initialization

Prior to the first pass thru FINDCDUW in any powered guidance phase, the routine INITCDUW must be called to initialize inputs to FINDCDUW. INITCDUW does the following:

1) OGABIAS = 0

2) $\underline{u}_{FV}$ = 1, 0, 0

3) $\underline{u}_{WDP}$ = 1, 0, 0

If the user wishes an outer gimbal angle bias, he must subsequently set OGABIAS.

5.3.7.3.2 Each Pass Initialization

The initialization which must be done each pass thru FINDCDUW amounts to acquisition and preparation of constants and input data (See Fig. 3.7-2). These are:

1. θ_{XM}, θ_{YM}, θ_{ZM}, the maximum angular changes are selected depending upon whether or not the Command and Service Modules are docked.

2. FLAGOODW is initialized to 0 or 1 according to whether X axis override is permitted or inhibited. This flag indicates whether the unit window command is good for control about the X axis.

3. OVFIND, the interpretive overflow indicator, is set to zero.

4. If the switches on the LM console indicate FGNCS MODE and AUTO MODE the CDUDs are fetched for the thrust direction filter, otherwise the CDUs are fetched. Fetching the CDUs for the thrust direction filter when not in PGNCS MODE maintains the thrust direction filter even when the autopilot does not maintain the CDUDs.

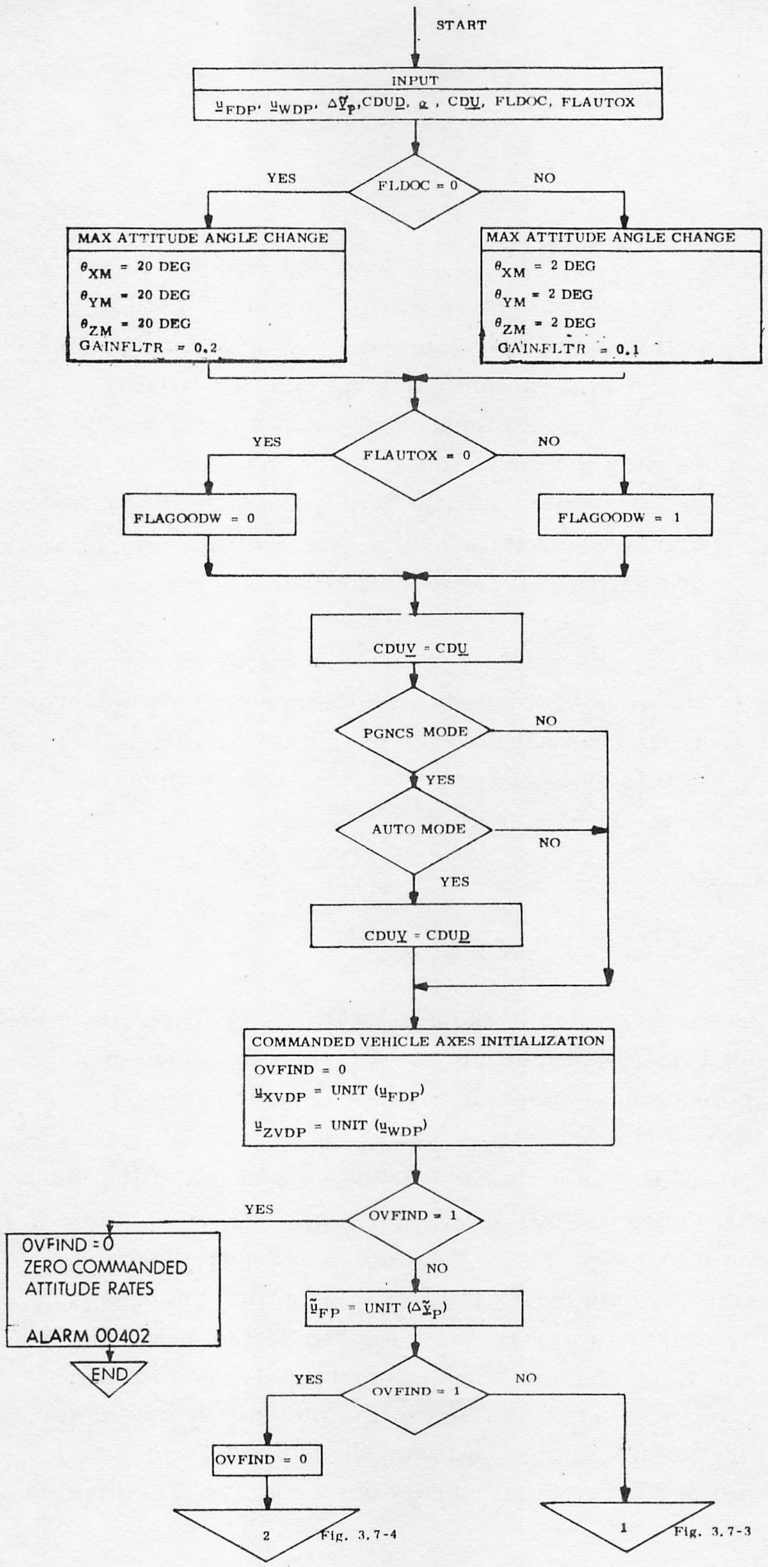


Figure 3.7-2 Each Pass Initialization

5. $\underline{u}_{XVDP}$ and $\underline{u}_{ZVDP}$, the X and Z axes of the commanded vehicle frame, are initialized to the unit of the thrust command $\underline{u}_{FDP}$ and to the unit of the window command $\underline{u}_{WDP}$. If unitizing either of these input commands fails either because a vector is too short or too long, the inputs are presumed unsatisfactory for attitude control, alarm 00402 is issued, attitude rate commands are zeroed and FINDCDUW returns to the calling program.

6. $\Delta\underline{\tilde{v}}_P$, the velocity increment vector measured by the accelerometers over the previous computation period, is unitized for the thrust direction filter. If this produces overflow, the thrust direction filter is bypassed.

5.3.7.3.3 Thrust Direction Filter

As the first step in the Thrust Direction Filter, Fig. 3.7-3, the unit measured thrust in platform coordinates is transformed to current vehicle coordinates to yield the unit measured thrust $\underline{\tilde{u}}_{FV}$. Then each of the two components, Y and Z, of the old unit thrust vector is subtracted from the corresponding component of the unit measured thrust, multiplied by the gain of the filter, GAINFLTR, and limited in magnitude to the maximum change which can be expected in one computation period, which yields the thrust direction changes Δu_{FYV} and Δu_{FZV}. The maximum changes are based on the slew rate of the trim gimbal. Each of these thrust direction changes is then added to the corresponding component of the old unit thrust vector and limited in magnitude to the maximum thrust offset which can be expected, yielding the new unit thrust vector $\underline{u}_{FV}$. The maximum

1

UNIT MEASURED THRUST, CURRENT VEHICLE COORDS

$$\underline{\tilde{u}}_{FV} = C_{PV}\,(\underline{C}DUV)\;\underline{\tilde{u}}_{FP}$$

THRUST DIRECTION CHANGES

$$\Delta u_{FYV} = (\text{LIMIT } .007)\;(\text{GAINFLTR } (\tilde{u}_{FYV} - u_{FYV}))$$

$$\Delta u_{FZV} = (\text{LIMIT } .007)\;(\text{GAINFLTR } (\tilde{u}_{FZV} - u_{FZV}))$$

UNIT THRUST VECTOR

$$u_{FYV} = (\text{LIMIT } .129)\;(u_{FYV} + \Delta u_{FYV})$$

$$u_{FZV} = (\text{LIMIT } .129)\;(u_{FZV} + \Delta u_{FZV})$$

2

Fig. 3.7-4

Fig. 3.7-3 Thrust Direction Filter.

thrust offset is based on the maximum displacement of the trim gimbal, the elastic deflection of the gimbal mount, the mounting alinement error, and the deflection of the thrust vector relative to the nozzle centerline.

5.3.7.3.4 Window Command Selection

In order to compute the axes of the commanded vehicle frame, it is first necessary to select a suitable unit window command (Fig. 3.7-4). The unit window command obtained from guidance may not be suitable because 1) X axis override may be permitted or 2) it may fail the test PARLTEST (a test for parallelism or anti-parallelism to the unit thrust command). If the unit window command cannot be used for either reason, it is necessary to fetch an alternate such that the resulting axes of the commanded vehicle frame will provide zero steady state thrust pointing error. Any vector lying in the +Z half of the LM symmetry plane will satisfy this requirement. Among these, the vector least likely to be parallel or anti-parallel to the unit thrust command is the Z axis ($\underline{u}_{ZBP}$). However, it is possible during a large maneuver that the Z axis also fails PARLTEST, and in this case a third alternative, the -X axis ($-\underline{u}_{XBP}$) is selected. Because the Z axis and the -X axis cannot both be parallel to the unit thrust command, PARLTEST is omitted for -X. The Z and -X axes selected are of the <u>body</u> and <u>not</u> the <u>vehicle</u> frame, simply to save program and time, as the Service Routine provides the body axes for all users. Using a body axis for window pointing causes no stability problem because any time a body axis is used, FLAGOODW is set to zero so that the attitude angle change about the X axis will also be set to zero. Thus, actual spacecraft motion about the X axis produces through FINDCDUW no attitude increment command about the X axis and greatly diminished commands about the Y and Z axes.

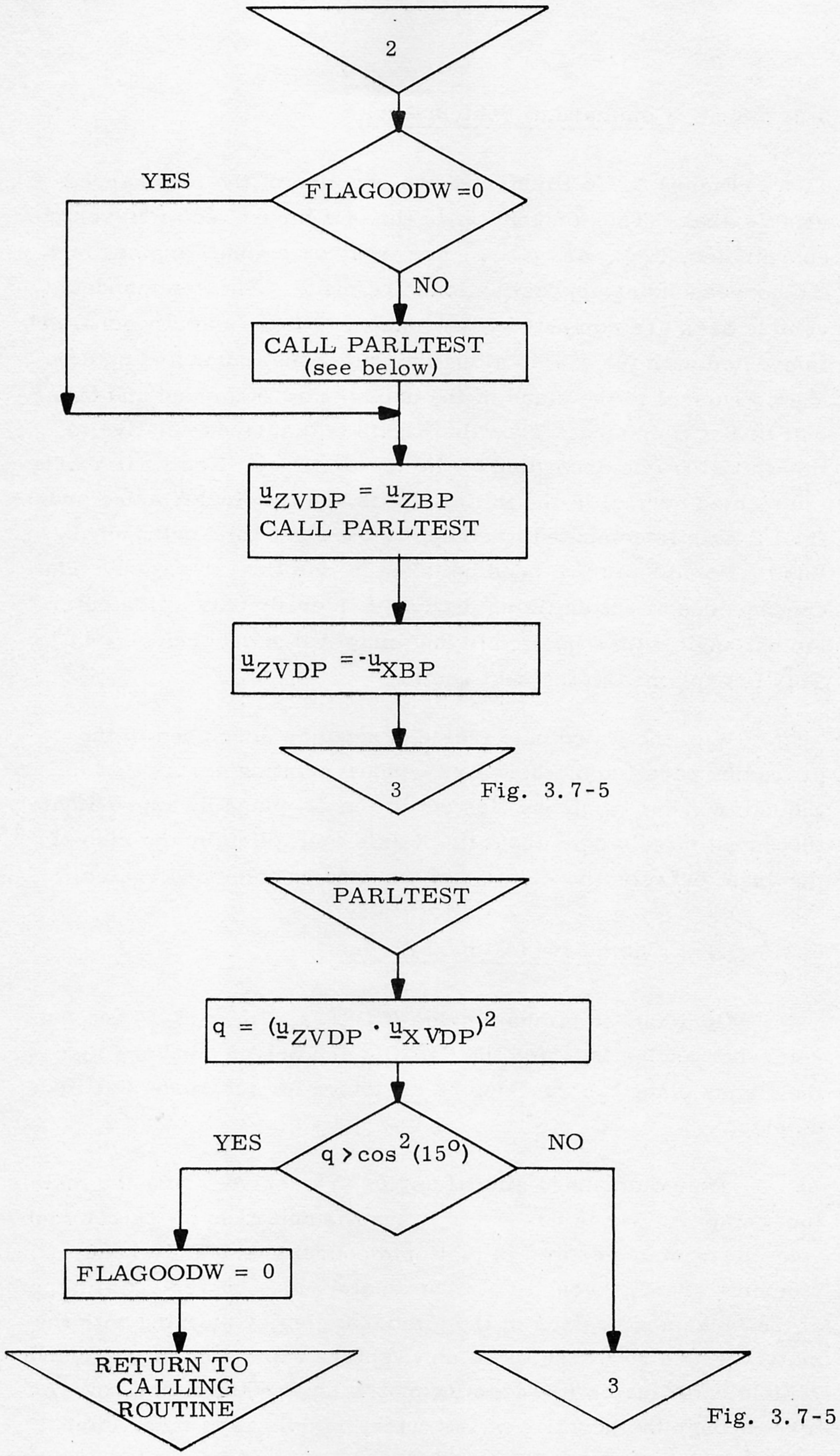


Fig. 3.7-4 Window Command Selection.

5.3.7.3.5 Commanded Vehicle Axes

Figure 3.7-5 illustrates the erection of the commanded vehicle axes. The components of the unit thrust vector in vehicle coordinates, u_{FYV} and u_{FZV}, are treated as small angles, and the corresponding approximations are made. The commanded vehicle axes are computed in two steps. First, an orthogonal set is erected with the X axis along the unit thrust command and the Y axis normal to the plane of the unit window command and the unit thrust command. Then the X axis is displaced relative to this initial frame according to the thrust offset. The final Y axis is oriented normal to the initial Z axis and the final X axis, and final Z axis is computed to complete the right hand orthogonal triad. No unit vectors need be taken in this final iteration. The consequence of not unitizing produces an error only in the outer gimbal angle with a negligible maximum value occurring at 45°. This is explained in the next section.

With the coordinate frame erected as described in the preceding paragraph, there is a window pointing error; that is, the unit window command lies out of the ZX plane by approximately the thrust offset angle about the X axis multiplied by the sine of the angle between the Z axis and the window command vector.

5.3.7.3.6 Commanded Gimbal Angles

Commanded gimbal angles (CDUC's, Fig. 3.7-6) are the reference angles to which the CDUD's are driven, and are also the angles placed in the Noun 22 registers for the Mode 2 attitude display.

The commanded gimbal angles are derived from the matrix just computed whose row vectors are the unit axes of the commanded vehicle frame expressed in platform coordinates, and whose elements are C_0 through C_8. This matrix may be derived in terms of the sines and cosines of the gimbal angles by starting with the matrix whose rows are these unit vectors expressed in commanded vehicle coordinates (the identity matrix) and rotating the row vectors through the negative of the outer, middle, and inner gimbal angles in that order. The resulting matrix may be used to identify the elements of interest of the matrix of the preceding paragraph as follows.

3

COMMANDED VEHICLE AXES, INITIAL ITERATION

$$\underline{u}_{YVDP} = \mathrm{UNIT}\,(\underline{u}_{ZVDP} \times \underline{u}_{XVDP})$$

$$\underline{u}_{ZVDP} = \underline{u}_{XVDP} \times \underline{u}_{YVDP}$$

X AXIS CORRECTION FOR THRUST OFFSET

$$\underline{u}_{XVDP} = \mathrm{UNIT}\,(\underline{u}_{XVDP} - u_{FYV}\,\underline{u}_{YVDP} - u_{FZV}\,\underline{u}_{ZVDP})$$

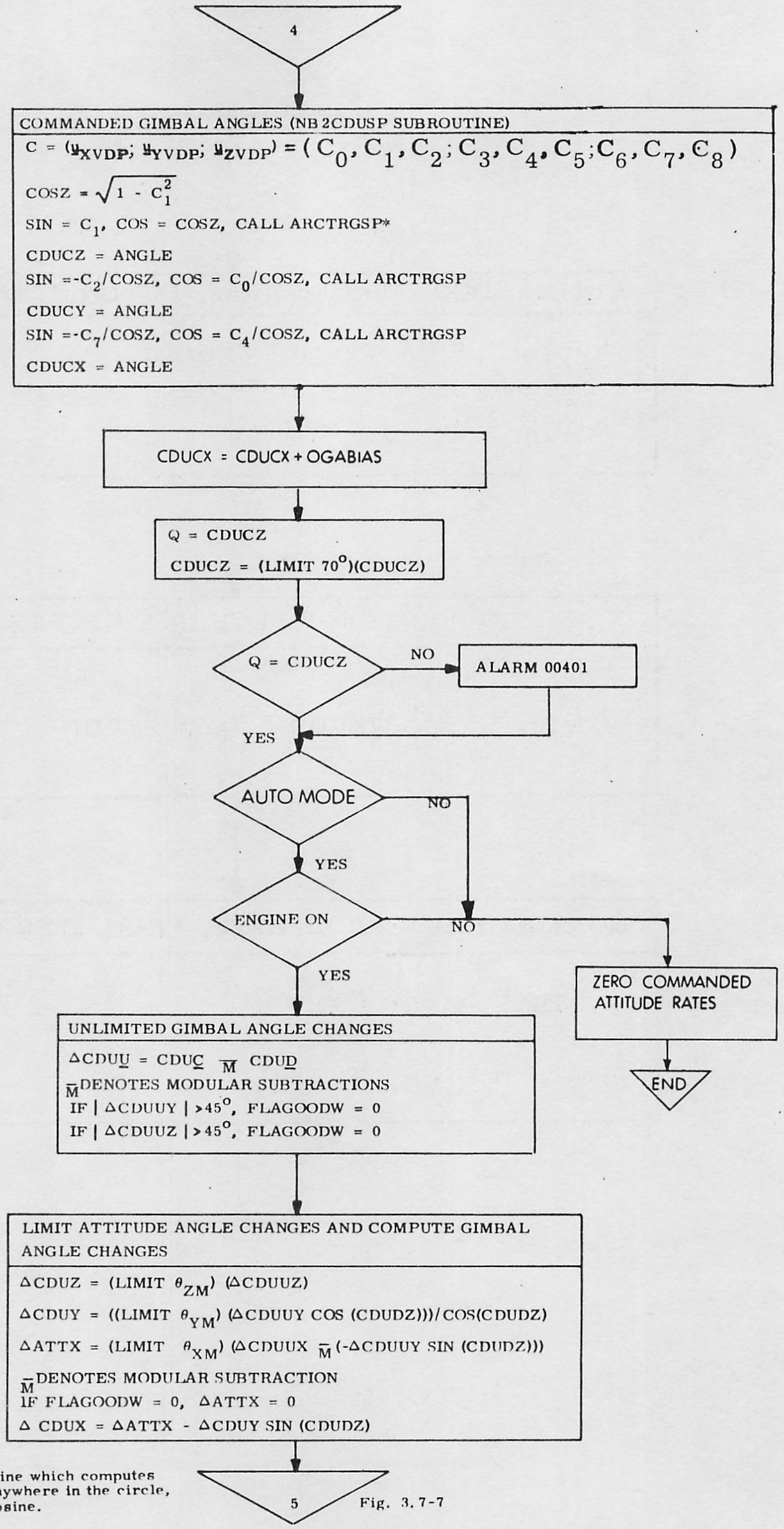
COMMANDED VEHICLE AXES, FINAL ITERATION

$$\underline{u}_{YVDP} = \underline{u}_{ZVDP} \times \underline{u}_{XVDP}$$

$$\underline{u}_{ZVDP} = \underline{u}_{XVDP} \times \underline{u}_{YVDP}$$

4 Fig. 3.7-6

Fig. 3.7-5 Commanded Vehicle Axes.



*ARCTRGSP is a subroutine which computes an unambiguous angle anywhere in the circle, given the sine and the cosine.

Figure 3.7-6 Commanded Gimbal Angles and Gimbal Angle Changes

$$C_0 = \cos(\text{CDUCZ}) \quad \cos(\text{CDUCY})$$

$$C_1 = \sin(\text{CDUCZ})$$

$$C_2 = -\cos(\text{CDUCZ}) \quad \sin(\text{CDUCY})$$

$$C_4 = \cos(\text{CDUCZ}) \quad \cos(\text{CDUCX})$$

$$C_7 = -\cos(\text{CDUCZ}) \quad \sin(\text{CDUCX})$$

The subroutine NB2CDUSP extracts the commanded gimbal angles, $\underline{\text{CDUC}}$, by exploiting these relationships. $|\text{CDUCZ}|$ is $< 90^\circ$, even before limiting.

ARCTRGSP is a subroutine of the subroutine NB2CDUSP. ARCTRGSP extracts angles given the sine and the cosine. To this end, ARCTRGSP extracts the arcsine of whichever given argument is smaller in magnitude, and uses only the sign of the larger argument for determining the quadrant.

Not unitizing the vectors in the final iteration for the commanded vehicle axes, as noted in the preceding section, reduces slightly the magnitude of the arguments used to extract the outer gimbal angle. Because ARCTRGSP extracts the arcsine of whichever argument is smaller in magnitude, a "unit" vector of length .992 would produce a maximum outer gimbal angle error (at 45°) of $.458^\circ$. A more probable length is .99966 which would produce a maximum error of only $.0197^\circ$.

If the mode is not AUTO, or if the engine is off, a STOPRATE is issued and control is returned to the caller of FINDCDUW.

The unlimited gimbal angle changes are computed by modular subtracting the autopilot's CDUD's from the CDUC's. The modular subtractions yield the smallest angular differences, e.g. $177^\circ - (-178^\circ) = -5^\circ$, not $+355^\circ$. If either the Y axis or the Z axis unlimited gimbal angle change exceeds 45° in magnitude, FLAGOODW is set to zero. This is to prevent false aborts in attitude about the X axis as will be explained in the next paragraph.

Attitude angle changes, $\Delta\underline{\text{A}}\text{TT}$, are handled in a vehicle frame not previously described. The X axis of the frame is identical to the X axis of the current frame, the Z axis is along the middle gimbal axis, and the Y axis completes the orthogonal

right hand triad. Thus, this frame is the current vehicle frame rotated about the X axis to bring the Z axis into alinement with the middle gimbal axis. This frame is chosen for computational simplicity. The Z axis gimbal angle change ΔCDUZ is identical to the Z axis attitude angle change ΔATTZ. The unlimited Y axis gimbal angle change is multiplied by cos (CDUDZ) to yield the the unlimited Y axis attitude angle change, this is limited and divided by cos (CDUDZ) to yield the limited Y axis gimbal angle change ΔCDUY. The unlimited X axis attitude angle change is computed using the unlimited X and Y axes gimbal angle changes. Note that the modular subtraction always produces a result under 180° in magnitude. This computation produces the smallest angle total rotation required about the X axis, although the validity diminishes as the gimbal angle changes become large. Under certain conditions the X axis attitude angle change computed will be of opposite sign compared to the succeeding pass. To prevent starting such a maneuver in the wrong direction about the X axis, ΔATTX is set to zero whenever either the Y axis or the Z axis unlimited gimbal angle change exceeds a limit value. Finally the limited X axis gimbal angle change, ΔCDUX, is computed using the limited attitude change, ΔATTX, and the limited gimbal angle change ΔCDUY.

Because orthogonal components of the attitude rate are limited independently, the length of the attitude rate vector is allowed to exceed the limit value for any one component.

5.3.7.3.7 Autopilot Data

With all limiting completed, it is now possible to compute a consistent set of input data for the digital autopilot using the $\Delta CD\underline{U}$'s (Fig. 3.7-7). The $\Delta CD\underline{U}$'s are divided by the computation period and the resulting gimbal angle rates are transformed to yield the commanded attitude rates $\underline{\omega}_{DV}$. The gimbal angle changes are multiplied by the ratio of the control sample period to the computation period to yield the gimbal angle increments $\delta CD\underline{U}$. Finally, the commanded attitude lag angles $\underline{\phi}$, which account for the angles by which the attitude will lag behind a ramp angular command due to the finite angular accelerations available, are computed using the available angular accelerations $\underline{\alpha}$ (the one jet accelerations supplied by the autopilot, doubled) and then individually magnitude limited.

5

COMMANDED ATTITUDE RATES

$$\underline{\omega}_{DV} = \begin{bmatrix} 1 & \sin(CDUDZ) & 0 \\ 0 & \cos(CDUDZ)\cos(CDUDX) & \sin(CDUDX) \\ 0 & -\cos(CDUDZ)\sin(CDUDX) & \cos(CDUDX) \end{bmatrix} \Delta \underline{C}DU/2$$

GIMBAL ANGLE INCREMENTS AND COMMANDED ATTITUDE LAG ANGLES

$$\delta\ \underline{C}DU = \frac{0.1}{2}\ \Delta \underline{C}DU$$

$$\phi_X = (\text{LIMIT } 10^{\circ})\ (\omega_{DXV} \left| \omega_{DXV} \right| / 2\ \alpha_X)$$

$$\phi_Y = (\text{LIMIT } 10^{\circ})\ (\omega_{DYV} \left| \omega_{DYV} \right| / 2\ \alpha_Y)$$

$$\phi_Z = (\text{LIMIT } 10^{\circ})\ (\omega_{DZV} \left| \omega_{DZV} \right| / 2\ \alpha_Z)$$

END

Fig. 3.7-7 Autopilot Data.

5.3.8 MIDTOAVE ROUTINE

The purpose of the MIDTOAVE Routine (R41) is to insure that the extrapolated state used to initialize the Average-G computations is consistent with the time at which the Average-G equations are actually turned on. The information flow diagram for this routine is presented in Fig. 3.8-1.

Before this routine is entered the ignition-time flag (FLTIG) is set. If the calling program is the Thrust Monitor Program (P47), FLTIG has been set to 1; otherwise FLTIG = 0. The time at which the Average-G equations are to be turned on (t_{AG}) is normally set to 29.9 seconds prior to ignition time (t_{IG}). If P47 is the calling program, t_{IG} is not defined and the Average-G equations will be turned on as soon as possible. In the powered landing maneuver t_{IG} is obtained from the Guidance and Control Routine.

Initially, a state vector ($\underline{r}'$, $\underline{v}'$) and its corresponding time (t') are available to the precision Coasting Integration Routine. This time will normally be several minutes before the computed ignition time (t_{IG}). The first step in the computation cycle is to check the ignition-time flag (FLTIG). A value FLTIG = 0 indicates that the ignition time has not as yet been slipped. If FLTIG = 0, a test is made to determine whether there are at least 20 seconds remaining before the Average-G turn-on time (t_{AG}). If there are less than 20 seconds remaining, the ignition-time flag (FLTIG) is set to unity and t_{AG} is advanced to 20 seconds beyond the present time (t). In addition, an alarm is sent to the crew to signal that the ignition time will be slipped. Note that the crew alarm is sent only once, since FLTIG = 1 on subsequent passes through the routine.

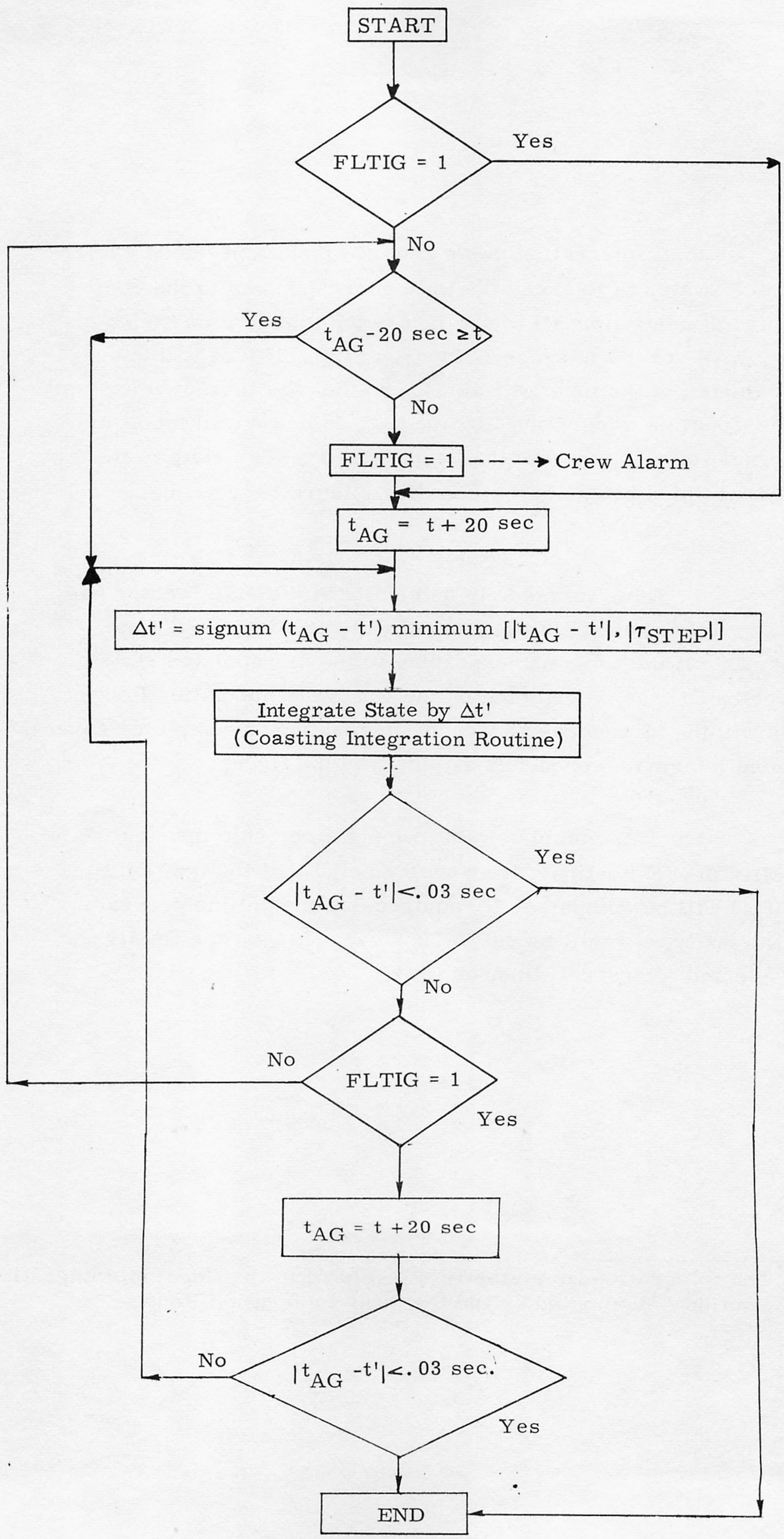


Figure 3.8-1. MIDTOAVE Routine

If integration is not finished from a previous cycle, the next step is to check the time corresponding to the computed state-vector (t') to see if it is within one time step (τ_{STEP})[†] of the Average-G turn-on time. If t' is within one time step of the turn-on time (t_{AG}), then the time step used in the Coasting Integration Routine ($\Delta t'$) is decreased from the initial value (τ_{STEP}) to the value $t_{AG} - t'$. The state is then extrapolated by $\Delta t'$ in the Coasting Integration Routine.

A test is made to determine if the state-vector has been integrated all the way to the desired Average-G time (t_{AG}). If the time corresponding to the extrapolated state-vector (t'), as determined by the Coasting Integration Routine, is within .03 seconds of t_{AG} the integration is complete; otherwise integration proceeds as shown in the figure.

This completes the computation cycle through the MIDTOAVE Routine. It should be noted that the ignition time (t_{IG}) will be computed, if required, by the calling program. Normally, t_{IG} will be set to 29.9 seconds past the finally-selected Average-G turn-on time.

[†] The integration time step (τ_{STEP}) used in the Coasting Integration Routine is computed by the Coasting Integration Routine.

5.4 TARGETING ROUTINES

5.4.1 GENERAL COMMENTS

The objectives of the targeting routines presented in this section are to provide the LGC capability to determine some of the required input target parameters and set control modes for the various powered flight guidance routines of Section 5.3. The LGC targeting capability represented by the programs described in the following subsections is limited to rendezvous maneuvers and aborts initiated during the powered lunar landing maneuver. These LGC targeting routines are presented in the following subsections.

5.4.2 Rendezvous Targeting

These routines include the following rendezvous maneuver phases:

1. Pre-CSI
2. Pre-CDH
3. Pre-TPI
4. Rendezvous Midcourse Correction

5.4.3 Abort Targeting from Lunar Landing

Initial landing time computations and required descent orbit injection and landing maneuver target parameter computations must be determined by the RTCC. The abort case involving transearth injection SPS backup must also be targeted by the RTCC.

All LGC targeting programs use the Basic Reference Coordinate System defined in Section 5.1.4.1. The basic input parameters required by the targeting routines of this section are the vehicle state vector estimates determined by the navigation programs of Section 5.2. All the Rendezvous Targeting Routines of Section 5.4.2 can be used in either earth or lunar orbit.

5.4.2 RENDEZVOUS TARGETING

5.4.2.1 General

The rendezvous Targeting Programs P-32, P-33, P-34 and P-35 in the LGC are based on the Concentric Flight Plan rendezvous scheme. This scheme illustrated in Fig. 4.2-1 employs an active vehicle to ideally make three impulsive maneuvers to establish a rendezvous intercept trajectory with a passive vehicle in an approximately coplanar orbit. Sufficient constraints are imposed to allow only one degree of freedom in choosing the final rendezvous profile. These constraints include the timing of both the first and third maneuvers, the direction of the first maneuver, a unique formulation for the second maneuver and the relative geometry of the two vehicles at the final maneuver point. The second maneuver results in the active vehicle orbit becoming "coelliptic" or "concentric" with the passive vehicle orbit, resulting in the differential altitude between the orbits being approximately constant. The following names have been given to the basic maneuvers of the Concentric Flight Plan rendezvous profile:

1) Coelliptic Sequence Initiation (CSI),

2) Constant Differential Altitude (CDH),

3) Transfer Phase Initiation (TPI).

A rendezvous midcourse correction program is used in addition to the above three primary rendezvous programs to maintain an intercept trajectory from the TPI maneuver until the manual terminal rendezvous is started.

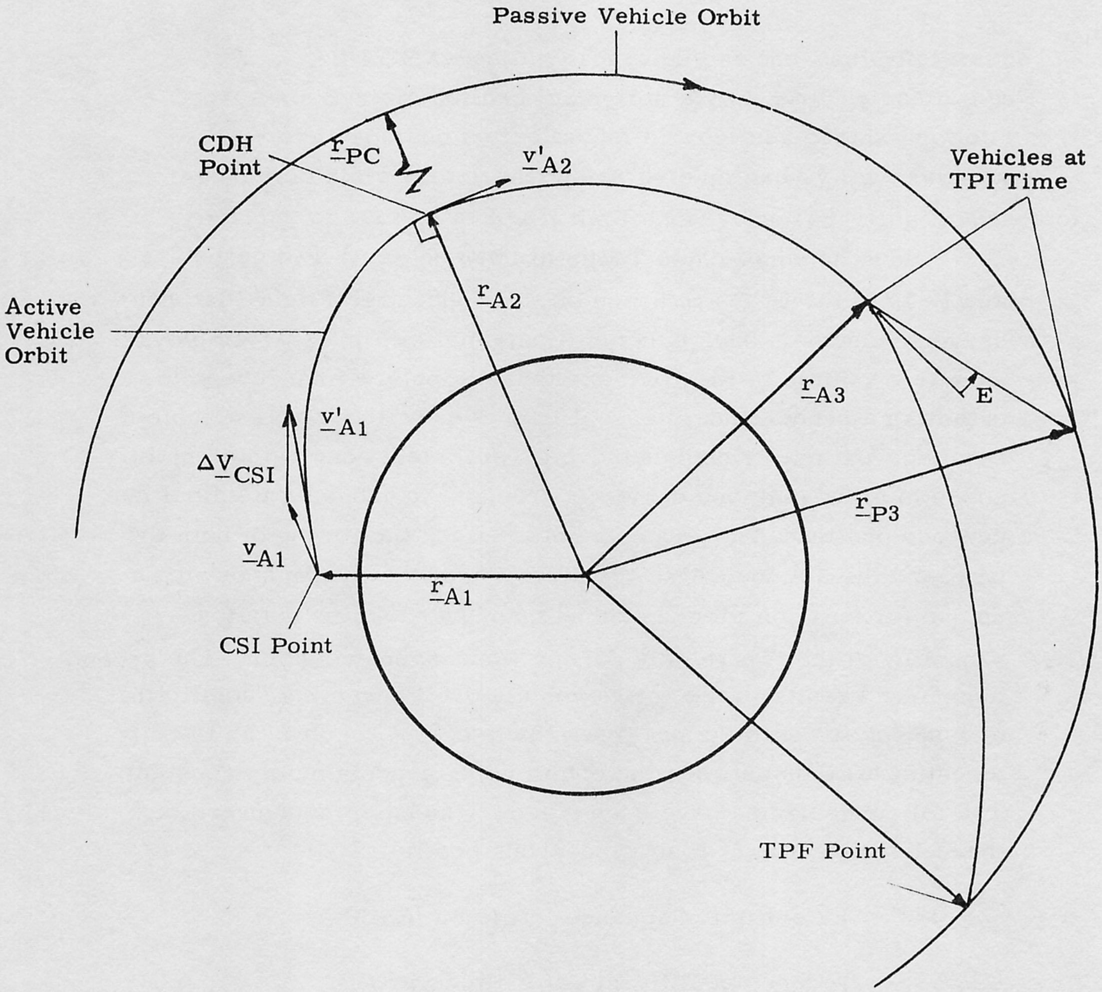


Figure 4.2-1 Concentric Rendezvous Profile

The LGC Rendezvous Targeting Programs consist of the following programs:

1) P-32 Pre-CSI Maneuver Program. This program is called by the astronaut any time before the Co-elliptic Sequence Initiation (CSI) maneuver is performed.

2) P-33 Pre-CDH Maneuver Program. This program is called between the CSI and Constant Differential Altitude (CDH) maneuvers.

3) P-34 Pre-TPI Maneuver Program. This program is called between the CDH and Transfer Phase Initiation (TPI) maneuvers.

4) P-35 Rendezvous Midcourse Maneuver Program. This program is called after the TPI maneuver has been performed.

Each program is designed to select either the LM or CSM as the active vehicle depending on the program number selected by the astronaut. All program operations in Section 5.4.2 are independent of which vehicle is chosen as the active vehicle.

A LM direct rendezvous capability is possible using the last two maneuvers in the above list.

All computations performed in the Concentric Rendezvous Program are in the Basic Reference Coordinate System (Section 5.1.4.1) and use the latest state vectors obtained from the Navigation Routines (Sections 5.2 and 5.3.2).

The pre-thrust programs P-32 and P-33 use External ΔV guidance (Section 5.3.3.3.1) and P-34 and P-35 use the Lambert targeting concept. There are 2 options available for obtaining the aim point used in generating the Lambert ΔV solution: If the number of offsets (N) is set to zero, the target orbit is advanced through a ωt in P34, or the appropriate time of flight in P35, by use of the Kepler (conic) routine. Lambert solution is then generated to this aim point.

The other option ($N>0$) employs N precision offsets to generate the velocity correction required. (N usually = 2). The target vehicle orbit is advanced the appropriate time of flight on a precision trajectory. The precision offset concept is then used to generate the ΔV solution. All four of the above programs use the Cross-Product Steering of Section 5.3.3.4.

An alarm code is displayed if the initial conditions along with the astronaut inputs fail to result in a satisfactory solution.

5.4.2.2 Pre-CSI Maneuver

This program, corresponding to Program P-32 (LM active) or P-72 (CSM active) of Section 4, computes the parameters associated with the CSI and CDH maneuvers. The astronaut inputs are:

1. Choice of active vehicle (P-32 LM, P-72 CSM).
2. Time t_1 of the CSI maneuver. If t_1 is equal to zero or negative, the program computes t_1 as the time of the next apoapsis, subject to astronaut modification.
3. Number N of the apsidal crossing. (If N = 1 and CDHASW is not equal to one, the CDH maneuver occurs when the active vehicle reaches its first apsidal point following the CSI maneuver, etc.)
4. Desired line of sight LOS angle E at the time of the TPI maneuver. (See Fig. 4.2-2 and Fig. 4.2-3).
5. Time t_3 of the TPI maneuver.
6. Selection of flag CDHASW to insure that CDH maneuver will occur at a time, measured from the CSI time, equal to the post CSI maneuver orbital period multiplied by N/2. If this option is desired, CDHASW is set equal to 1.

The active vehicle state vector $\underline{r}_A$, $\underline{v}_A$, t_A and passive vehicle state vector $\underline{r}_P$, $\underline{v}_P$, t_P are available in the guidance computer.

The following constraints must be satisfied (see Fig. 4.2-1 and Fig. 4.2-2).

1. The CSI ΔV is applied in the horizontal plane of the active vehicle at the CSI time.

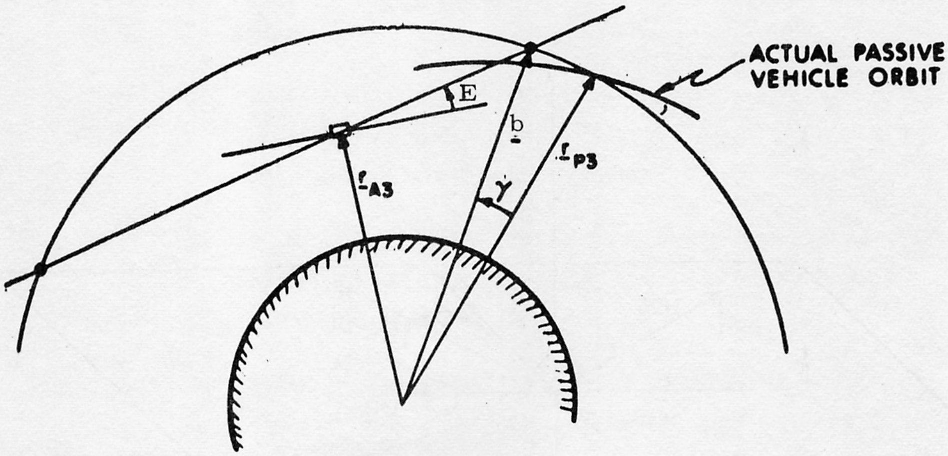


Fig. 4.2-2a **TPI geometry, active vehicle below**

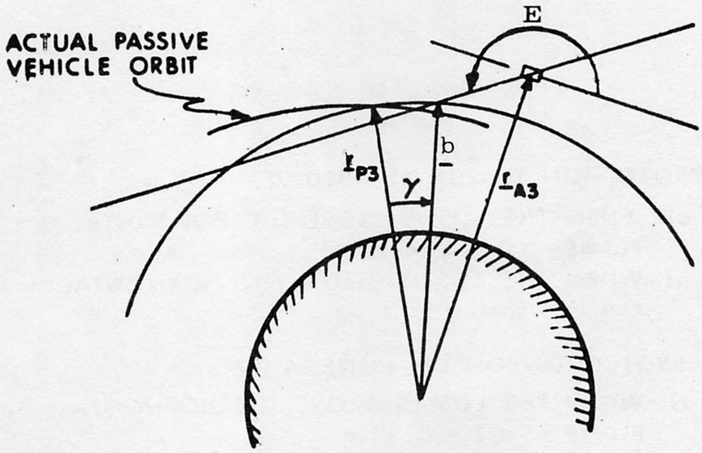
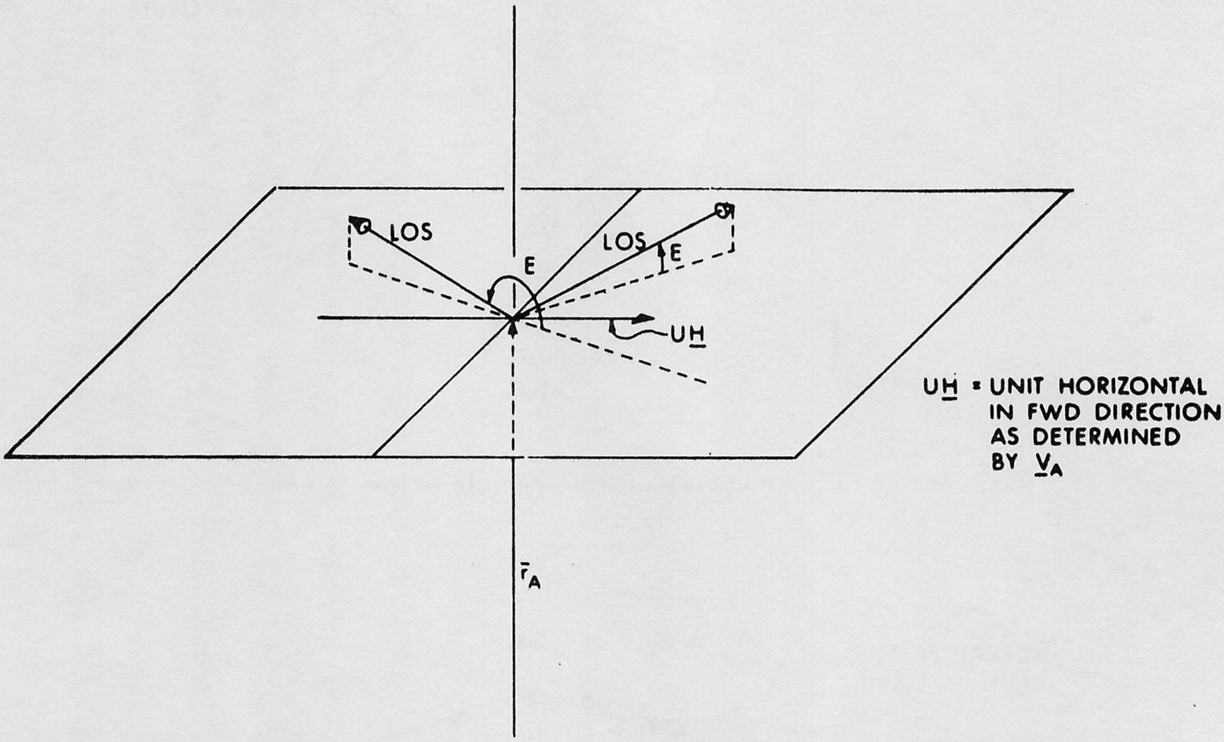


Figure 4.2-2b TPI geometry active vehicle above



1) IF THE LOS PROJECTION ON UH IS POSITIVE:

a) WHEN THE LOS IS ABOVE THE HORIZONTAL PLANE $0 < E < \pi/2$

b) WHEN THE LOS IS BELOW THE HORIZONTAL PLANE $3\pi/2 < E < 2\pi$

2) IF THE LOS PROJECTION ON UH IS NEGATIVE

a) WHEN THE LOS IS ABOVE THE HORIZONTAL PLANE $\pi/2 < E < \pi$

b) WHEN THE LOS IS BELOW THE HORIZONTAL PLANE $\pi < E < 3\pi/2$

Fig. 4.2-3 Definition **of Elevation Angle, E**

2. The semimajor axis and radial component of velocity of the active vehicle are such that the active and passive vehicles are in coelliptic orbits following the CDH maneuver.

3. The line of sight between the active vehicle and the passive vehicle at the TPI time forms the angle E with the horizontal plane of the active vehicle.

4. The time intervals between the CSI-CDH and CDH-TPI maneuvers are 10 minutes or greater.

5. After both the CSI and CDH maneuvers, the perigee altitude of the active vehicle orbit is greater than 35, 000 ft. for lunar orbits and 85 nm. for earth orbits.

The program solution, as illustrated in the logic flow diagram in Fig. 4.2-4, is based on conic trajectories and contains an iteration loop to select the CSI maneuver magnitude v_1.

After updating of the active and passive vehicle state vectors to the CSI time, the active vehicle state vector is rotated into the passive vehicle orbital plane resulting in coplanar orbits. The passive vehicle is then updated through the time $t_3 - t_1$ to the TPI point obtaining $\underline{r}_{P3}$, $\underline{v}_{P3}$. The initial guess of v_1, when added to the velocity of the active vehicle, results in the active vehicle attaining a radius equal to r_{P3} 180 degrees from the CSI point.

If CDHASW equals one, the eccentricity of the active vehicle orbit is less than 0.000488 or the vertical velocity at the CSI point is less than 7 ft/sec., the time t_2 of the CDH maneuver is set equal to the CSI time plus N times half the period t_P of the active vehicle. Otherwise, the angle ψ to the nearest perigee is computed and used in the Time-Theta Subroutine to find the corresponding time of flight Δt. This is then used to calculate t_2 based on the value of N. After verifying that t_3 is greater than t_2, both vehicles are updated to the CDH point.

The angle φ between the active and passive vehicles at the CDH point is calculated and used in the Time-Theta Routine to update the passive vehicle to a point radially above or below the active vehicle, obtaining $\underline{r}_{PC}$, $\underline{v}_{PC}$.

The semimajor axis a_A and the radial component of velocity v_{AV}, along with a radius, uniquely define the orbit of the active vehicle immediately following the CDH maneuver. The equations for a_A (based on the semimajor axis a_P of the passive vehicle) and v_{AV}, contained in Fig. 4.2-4, therefore specify the "coelliptic" CDH maneuver $\Delta \underline{v}_2$.

The active vehicle state vector is next updated to the TPI time. The unit vector which passes through the active vehicle position and is coincident with the desired TPI line of sight is given by $\underline{u}_L$. The position vector of the two points of intersection between the line of sight and a circle (with origin at the center of the attracting body) as shown in Fig. 4.2-2 is defined by:

$$\underline{b} = \underline{r}_{A3} + k\,\underline{u}_L \qquad (4.2.1)$$

where k normally assumes two values (see Fig. 4.2-4). Equating the magnitude of $\underline{b}$ to r_{P3} results in a quadratic expression for k

$$k^2 + 2\,k\,\underline{r}_{A3} \cdot \underline{u}_L + r_{A3}^{\;2} - r_{P3}^{\;2} = 0 \qquad (4.2.2)$$

If there are no real solutions to the above equation ($c_2 < 0$ in Fig. 4.2-4), the line of sight does not intersect the circle. If there is a real solution to the above equation, the k of minimum absolute value is chosen as the desirable TPI solution. The error γ is defined as the central angle between the position vector of the passive vehicle at the TPI point and the position vector $\underline{b}$ representing where the passive vehicle should be based on the active vehicle location. This error is driven to zero using a Newton-Raphson iteration loop with v_1 as the independent variable. The two initial values of v_1 are 10 ft/sec apart with the succeeding step size Δv_1 restricted to 200 ft/sec.

A second computational attempt is made if the first iteration attempt fails. v_1 is incremented in steps of 50 ft/sec. in the direction opposite to that initially taken in the first attempt until the error γ has undergone a sign change. The second attempt then starts with the latest value of v_1 and an adjacent point in the direction of the 50 ft/sec. steps.

Included in the Pre-CSI program are the following seven program checks and the corresponding Alarm Codes AC (see Fig. 4.2-4).

1) If on the first iteration there is no solution to the TPI geometry, the iteration cannot be established. The alarm code AC equals A in this case.

2. The iteration counter n exceeds 15. (AC = C)

3. Two succeeding iterations resulting in v_1 greater than 1000 ft/sec. (AC = B)

4. The altitude of perigee h_{P1} following the CSI maneuver is less than a minimum acceptable amount. (AC = D)

5. The altitude of perigee h_{P2} following the CDH maneuver is less than a minimum acceptable amount. (AC = E)

6. The time ΔT_2 between the CDH and CSI maneuver is less than 10 minutes. (AC = F)

7. The time ΔT_3 between the TPI and CDH maneuvers is less than 10 minutes. (AC = G)

For the first check above, a check failure results in an immediate program exit. The last four checks are made only if the iteration has succeeded. The alarm codes are set (excluding the first check) only if the second iteration attempt has been made.

The displays for the Pre-CSI maneuver program are:

1. Differential altitude ΔH_{CDH} at the CDH point. ΔH_{CDH} is positive when the active vehicle's altitude is less than the passive vehicle altitude when measured at the CDH point.

2. Time ΔT_2 between the CDH and CSI maneuvers displayed in minutes and seconds with hours deleted.

3. Time ΔT_3 between the TPI and CDH maneuvers displayed in minutes and seconds with hours deleted.

4. $\Delta \underline{V}_{CSI}(LV)$ in local vertical coordinates at CSI time.

5. $\Delta \underline{V}_{CDH}(LV)$ in local vertical coordinates at CDH time.

6. The alarm codes.

In the Pre-CSI program P-32, the astronaut can change the CSI computed maneuver by over writing the displayed maneuver velocity vector in local vertical coordinates, $\Delta\underline{V}_{CSI}$ (LV). In such a case the astronaut input then becomes the defined velocity-to-be-gained in the External ΔV Maneuver Guidance (Section 5.3.3.3.1) used to control the CSI maneuver. The over writing procedure is normally used to correct or minimize out-of-plane conditions which are not directly controlled by the CSI or CDH maneuvers computations such that minimum out of plane conditions exist at the TPI maneuver.

In addition, M the number of navigation measurements processed since the last maneuver, MGA the middle gimbal angle at the start of the maneuver assuming plus X acceleration, and TFI the time between the CSI maneuver and the present time are displayed.

The middle gimbal angle is computed with the following equation:

$$MGA = \sin^{-1}\left[\underline{Y}_{REF} \cdot \text{UNIT}\,(\Delta\underline{V})\right]$$

If $-90^\circ \leq MGA < 0^\circ$, MGA is redefined such that $270^\circ \leq MGA < 360^\circ$. $\Delta\underline{V}$ is the maneuver $\Delta\underline{V}$ in reference coordinates and $\underline{Y}_{REF}$ is the second row of the REFSMMAT matrix defined in Section 5.6.3.4.

5.4.2.3 Pre-CDH Maneuver

This program, corresponding to Program P-33 (LM active) or P-73 (CSM active) of Section 4, computes the parameters associated with the CDH maneuver.

The astronaut inputs to this program are:

1. Choice of active vehicle (P-33 LM, P-73 CSM)

2. Time t_2 of CDH maneuver

The active and passive vehicle's state vectors $\underline{r}_A$, $\underline{v}_A$, t_A, $\underline{r}_P$, $\underline{v}_P$, t_P are available in the guidance computer. The TPI elevation angle E and the TPI maneuver time t_3 are also available from the previous Pre-CSI program computations.

Figure 4.2-4, page 4 of 7, illustrates the program logic. The state vectors of both the active and passive vehicles are advanced to the CDH time using the Coasting Integration Routine. After rotating the active vehicle state vector into the plane of the passive vehicle, the CDH maneuver is calculated as in the Pre-CSI program. Following a precision update of the state vectors to time t_3, the Pre-TPI program is used to calculate the time t at which the specified elevation angle is attained. If the iteration was not successful, an alarm code is displayed. At this point, the astronaut can elect to recycle the program or to proceed with the calculation of the displays, which are:

1. $\Delta \underline{V}_{CDH}(LV)$
2. Differential altitude ΔH_{CDH} at the CDH point.
3. Time ΔT_3 between the calculated TPI time and the CDH time displayed in minutes and seconds with hours deleted.
4. Time ΔT_{TPI} between the calculated TPI time and the specified TPI time. (This number is positive if the new TPI time is later than that previously used.)

If the iteration did not converge, the last two displays reflect an unchanged TPI time. In addition, M, MGA and TFI are displayed as in the Pre-CSI program displays.

The astronaut can modify the CDH maneuver to minimize out-of-plane conditions by overwriting the displayed maneuver velocity vector, $\Delta \underline{V}_{CDH}(LV)$, as in the pre-CSI program. The out-of-plane information used in this process is determined by the astronaut with the aid of the Rendezvous Out-of-Plane Display Routine, R36, of Section 5.6.7.3.

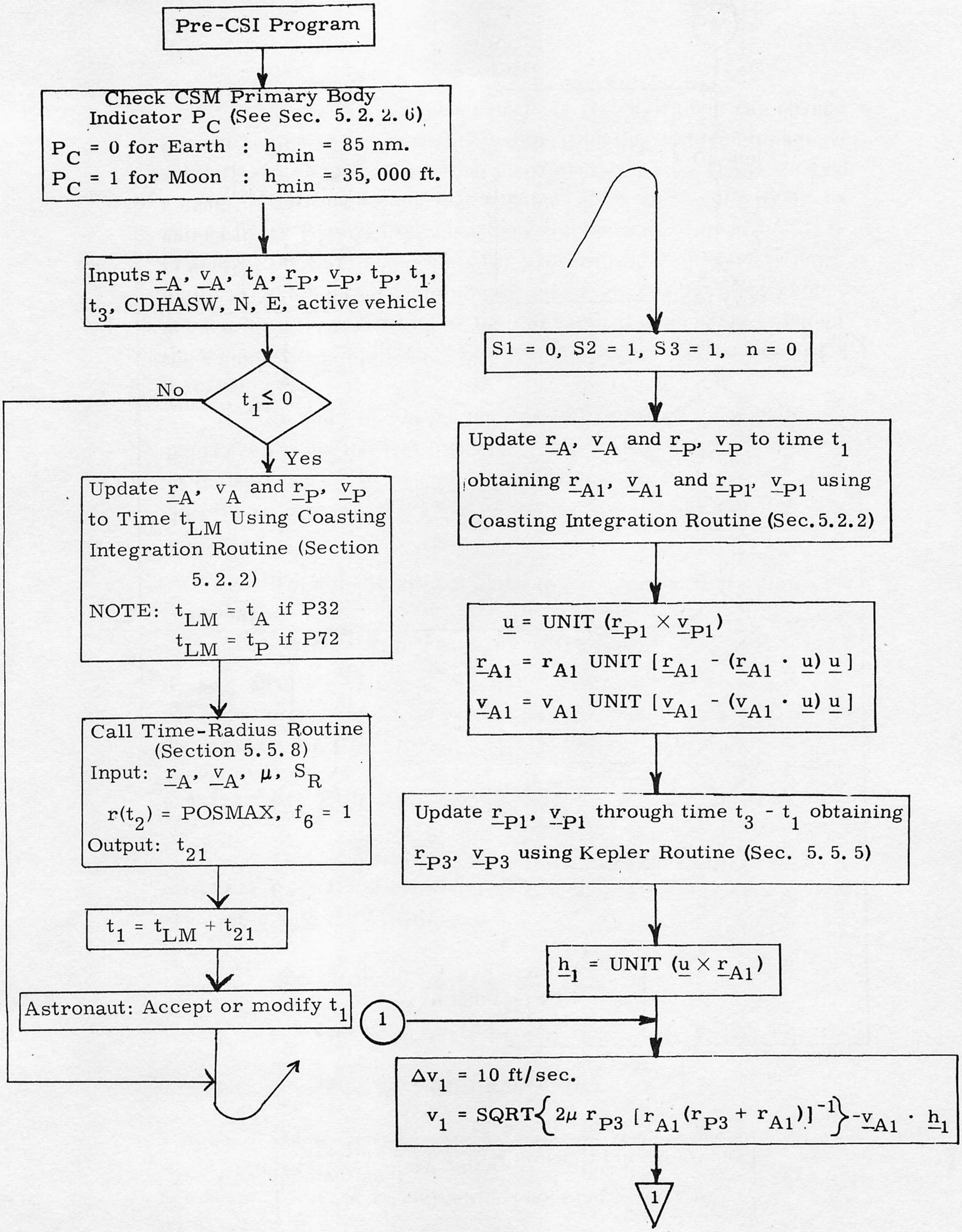


Figure 4.2-4 Pre-CSI-CDH Maneuver Program
(page 1 of 7)

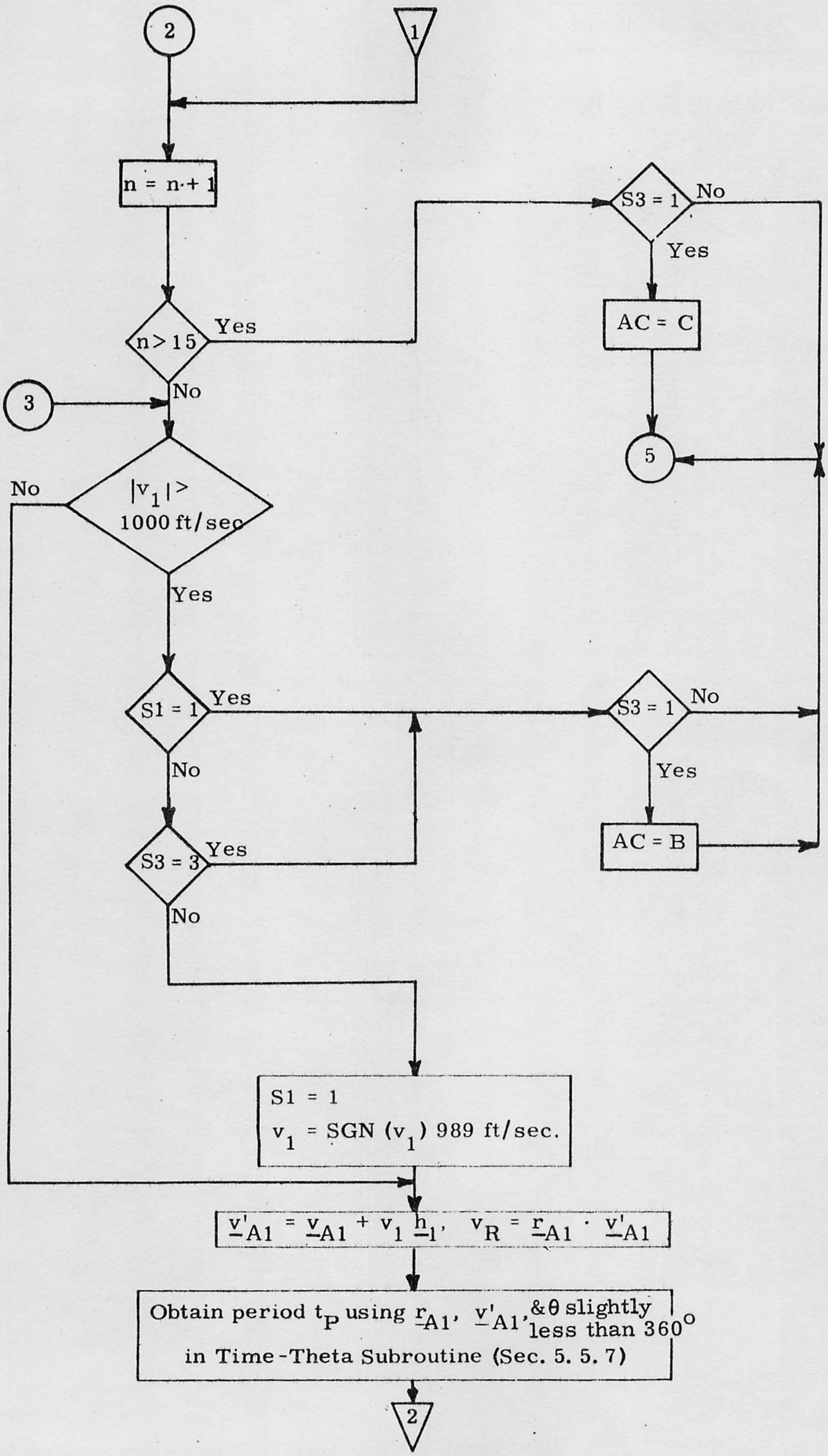


Figure 4.2-4 Pre-CSI-CDH Maneuver Program (page 2 of 7)

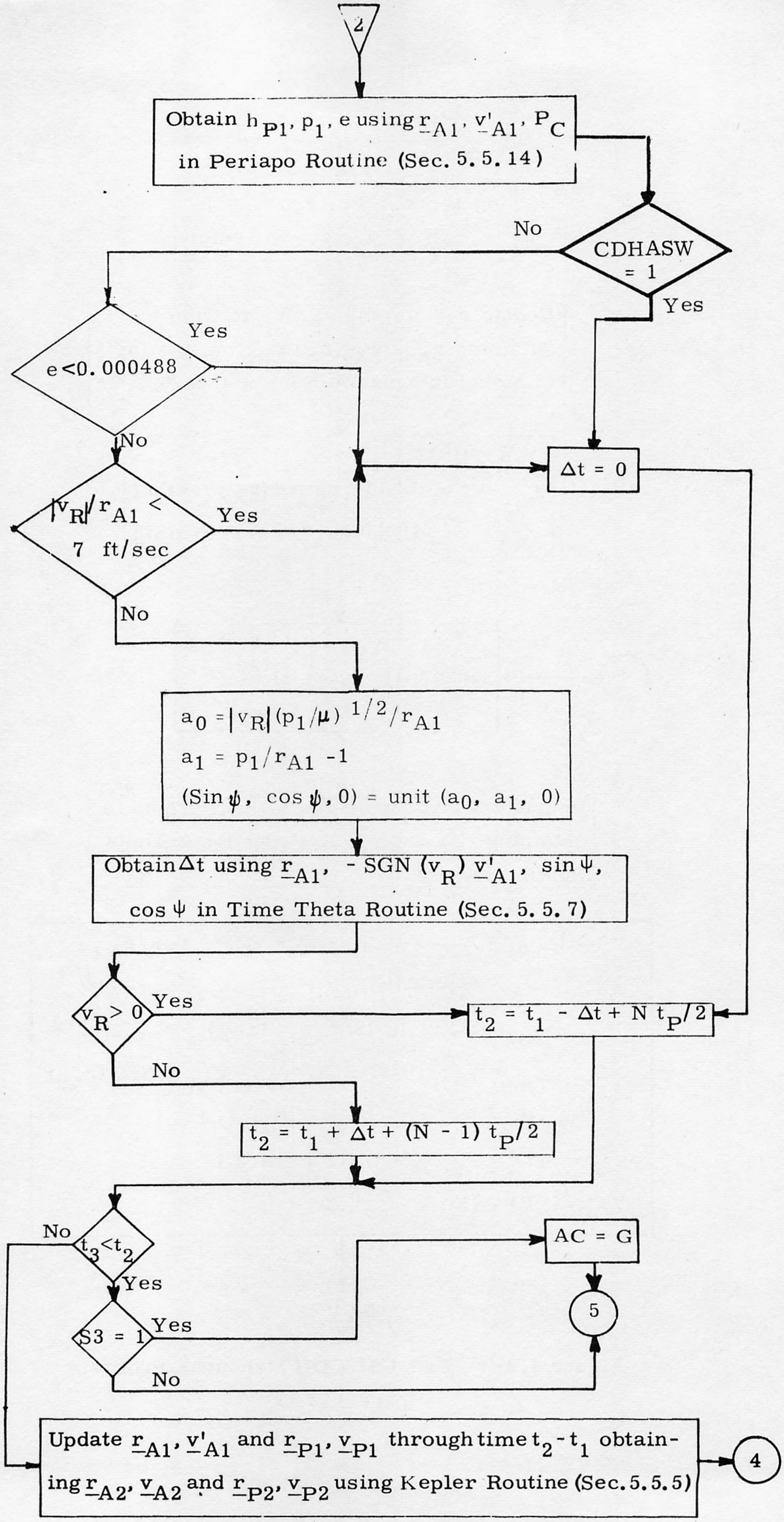


Figure 4.2-4 Pre-CSI-CDH Maneuver Program (page 3 of 7)

Pre-CDH Program

Inputs: $\underline{r}_A$, $\underline{v}_A$, $\underline{r}_P$, $\underline{v}_P$, E, t_2, t_3, active vehicle

Update $\underline{r}_A$, $\underline{v}_A$ and $\underline{r}_P$, $\underline{v}_P$ to time t_2 obtaining $\underline{r}_{A2}$, $\underline{v}_{A2}$ and $\underline{r}_{P2}$, $\underline{v}_{P2}$ using Coasting Integration Routine (Sec. 5.2.2)

$$\underline{u} = \text{UNIT}\ (\underline{r}_{P2} \times \underline{v}_{P2})$$
$$\underline{r}_{A2} = r_{A2}\ \text{UNIT}\ [\underline{r}_{A2} - (\underline{r}_{A2} \cdot \underline{u})\ \underline{u}]$$
$$\underline{v}_{A2} = v_{A2}\ \text{UNIT}\ [\underline{v}_{A2} - (\underline{v}_{A2} \cdot \underline{u})\ \underline{u}]$$

4 →

$$\cos\phi = \underline{r}_{A2} \cdot \underline{r}_{P2} / r_{A2}\, r_{P2}$$
$$\sin\phi = (1 - \cos^2\phi)^{1/2}$$
$$SW = \underline{u} \cdot (\underline{r}_{P2} \times \underline{r}_{A2})$$

Update $\underline{r}_{P2}$, $\underline{v}_{P2}$ with $\cos\phi$, SGN (SW)$\sin\phi$ obtaining $\underline{r}_{PC}$, $\underline{v}_{PC}$, a_P using Time-Theta Routine (Sec. 5.5.7)

$$\Delta H_{CDH} = r_{PC} - r_{A2},\quad v_{PV} = \underline{v}_{PC} \cdot \underline{r}_{A2}/r_{A2}$$
$$a_A = a_P - \Delta H_{CDH},\quad v_{AV} = v_{PV}\,(a_P/a_A)^{3/2}$$
$$v_{AH} = \text{SQRT}\ [\mu\,(2/r_{A2} - 1/a_A) - v_{AV}^{\ 2}]$$
$$\underline{v}'_{A2} = v_{AH}\ \text{UNIT}\ (\underline{u} \times \underline{r}_{A2}) + v_{AV}\ \text{UNIT}\ (\underline{r}_{A2})$$
$$\underline{\Delta v}_2 = \underline{v}_{A2}' - \underline{v}_{A2},\quad \underline{Z} = \text{UNIT}\ (-\underline{r}_{A2})$$
$$\underline{Y} = -\underline{u},\quad \underline{X} = \text{UNIT}\ (\underline{Y} \times \underline{Z})$$
$$\underline{\Delta V}_{CDH}(LV) = [\underline{X}\ \ \underline{Y}\ \ \underline{Z}]^T\ \underline{\Delta v}_2$$

3

Figure 4.2-4 Pre-CSI-CDH Maneuver Program (page 4 of 7)

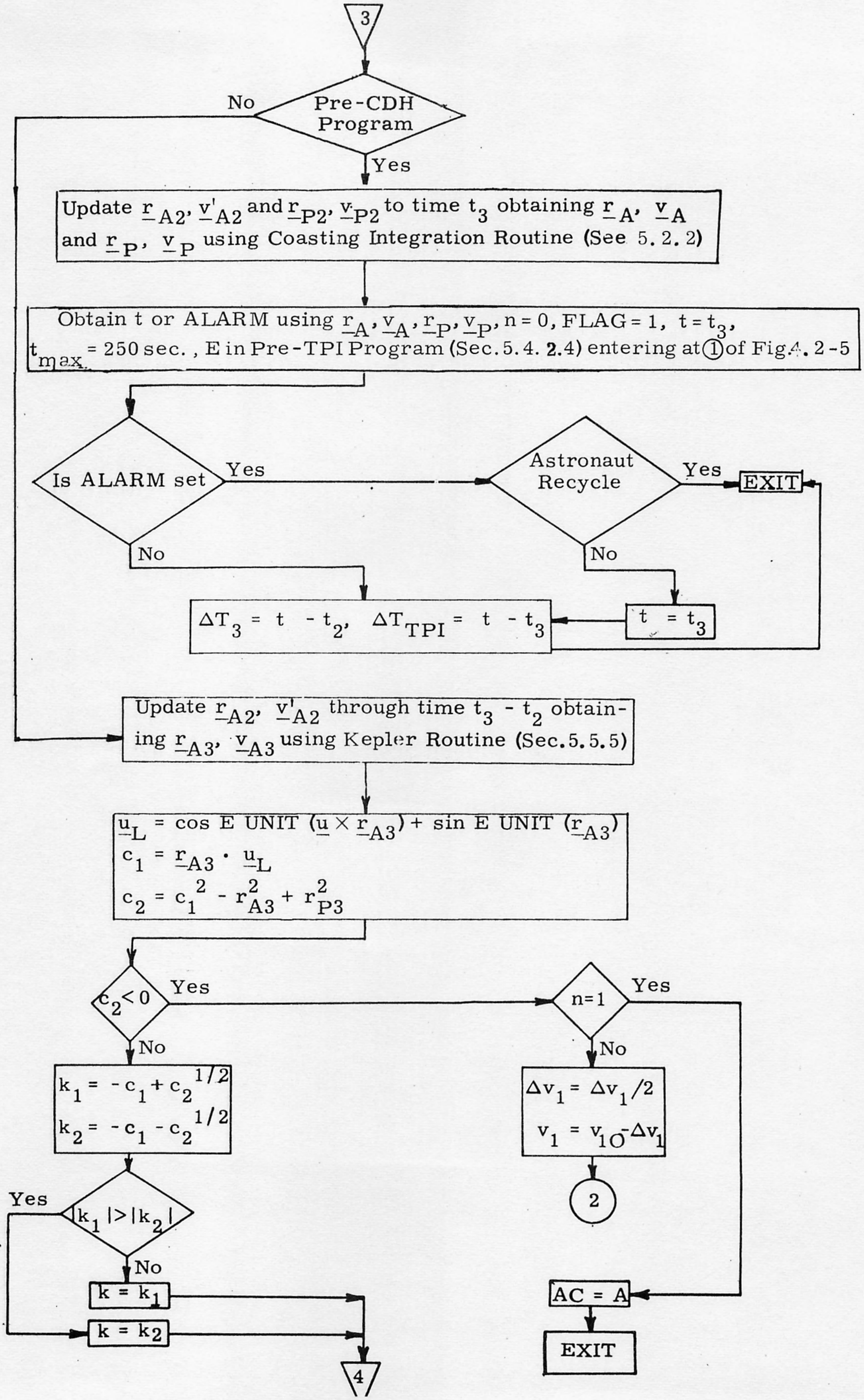


Figure 4.2-4 Pre-CSI-CDH Maneuver Program

(page 5 of 7)

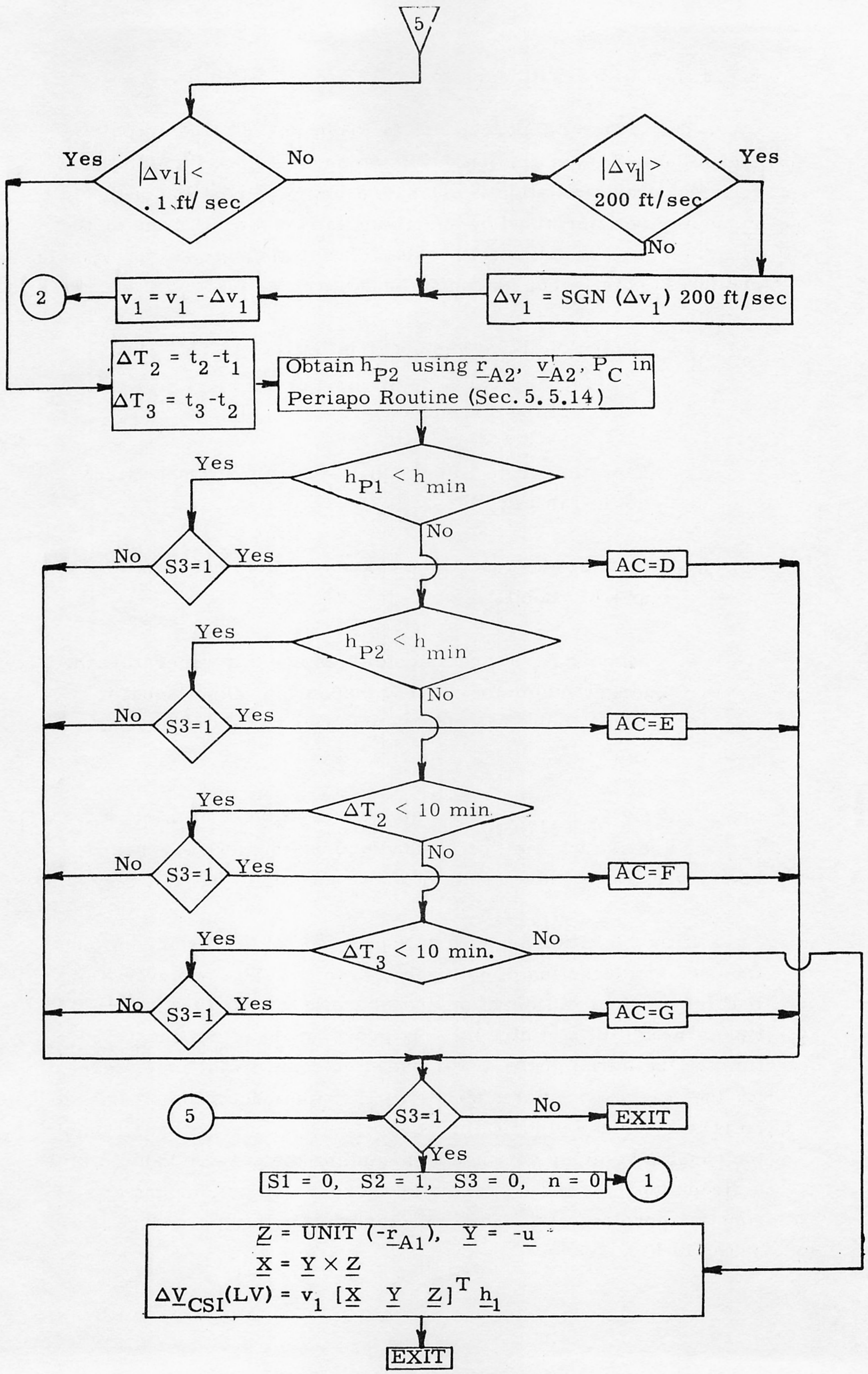


Figure 4.2-4 Pre-CSI-CDH Maneuver Program
(page 7 of 7)

5.4.2.4 Pre-TPI Maneuver

This program corresponds to Program P34 (LM active) or P74 (CSM active) of Section 4. Its objective is to establish the terminal phase initiation (TPI) maneuver. The position of the TPI maneuver is determined by specifying either the TPI time or the elevation angle which specifies the relative geometry of the vehicles at the TPI point. The astronaut inputs are:

1. Choice of active vehicle (P34 LM, P74 CSM)

2. Time t of the TPI maneuver

3. Elevation angle E (set equal to zero if t is specified) defined in Fig. 4.2-3

4. Central angle ωt of the passive vehicle between the TPF and TPI points

5. Number N_1 of precision offsets made in generating the target vector for the TPI maneuver. If set equal to zero, conic trajectories are used to generate the target vector.

The active $\underline{r}_{AI}$, $\underline{v}_{AI}$, t_{AI} and passive vehicle $\underline{r}_{PI}$, $\underline{v}_{PI}$, t_{PI} state vectors are available in the guidance computer. The program starts with a precision update of these vectors to the TPI time.

If the elevation angle is not specified, it is computed after the precision updating of both vehicles to the TPI maneuver time. If E is specified, an iteration procedure is initiated to find the TPI time at which E is attained. This procedure uses the input TPI time as the initial guess and is based on conic trajectories (except for the original precision update). The time correction δt is based on (1) the angular distance between $\underline{r}_P$ and the desired position of the passive vehicle, obtained by assuming the passive vehicle is in a circular orbit, and (2) assuming the vehicles are moving at a constant angular rate. The flow diagram for this procedure is shown in Fig. 4.2-5.

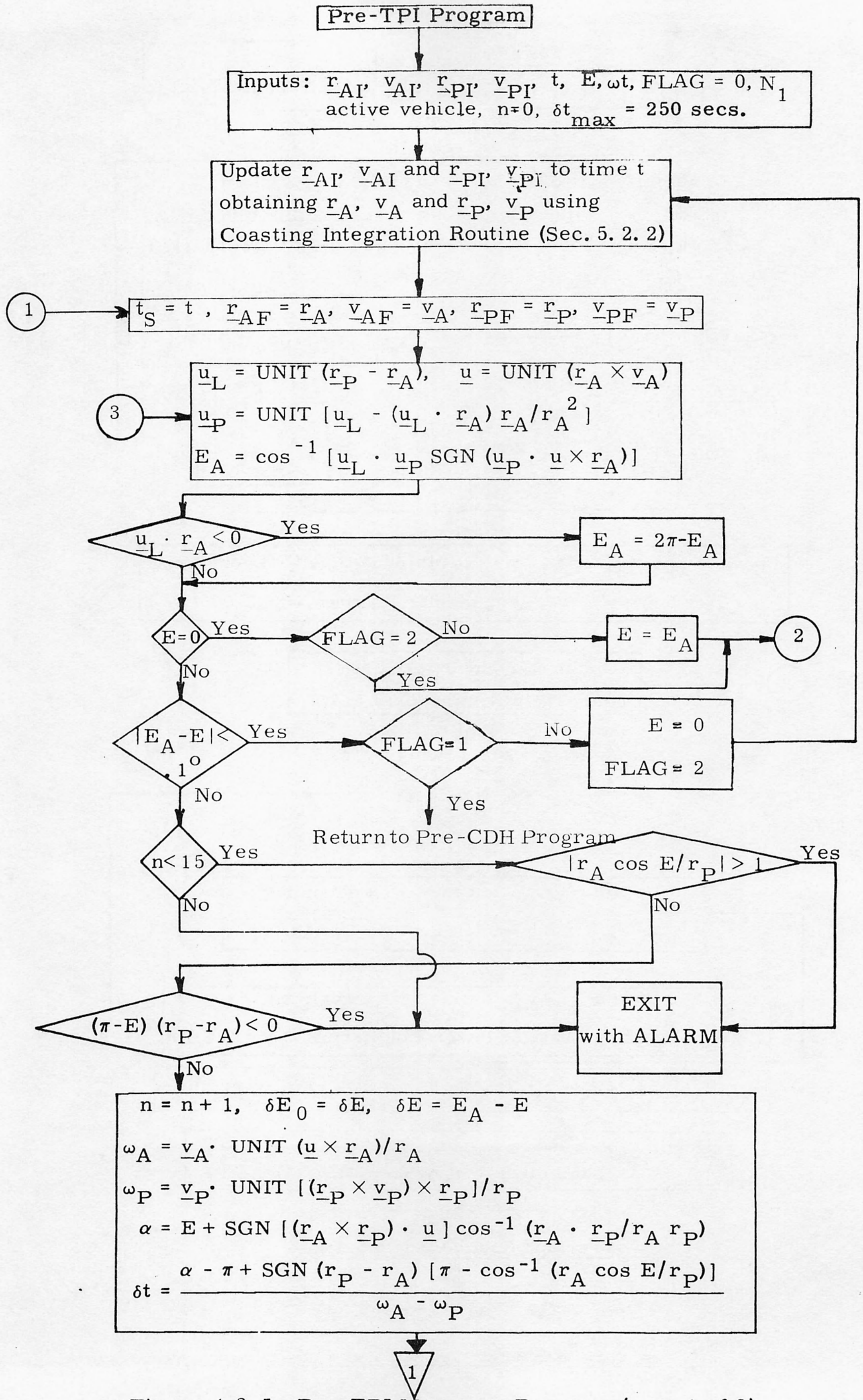


Figure 4.2-5 Pre-TPI Maneuver Program (page 1 of 3)

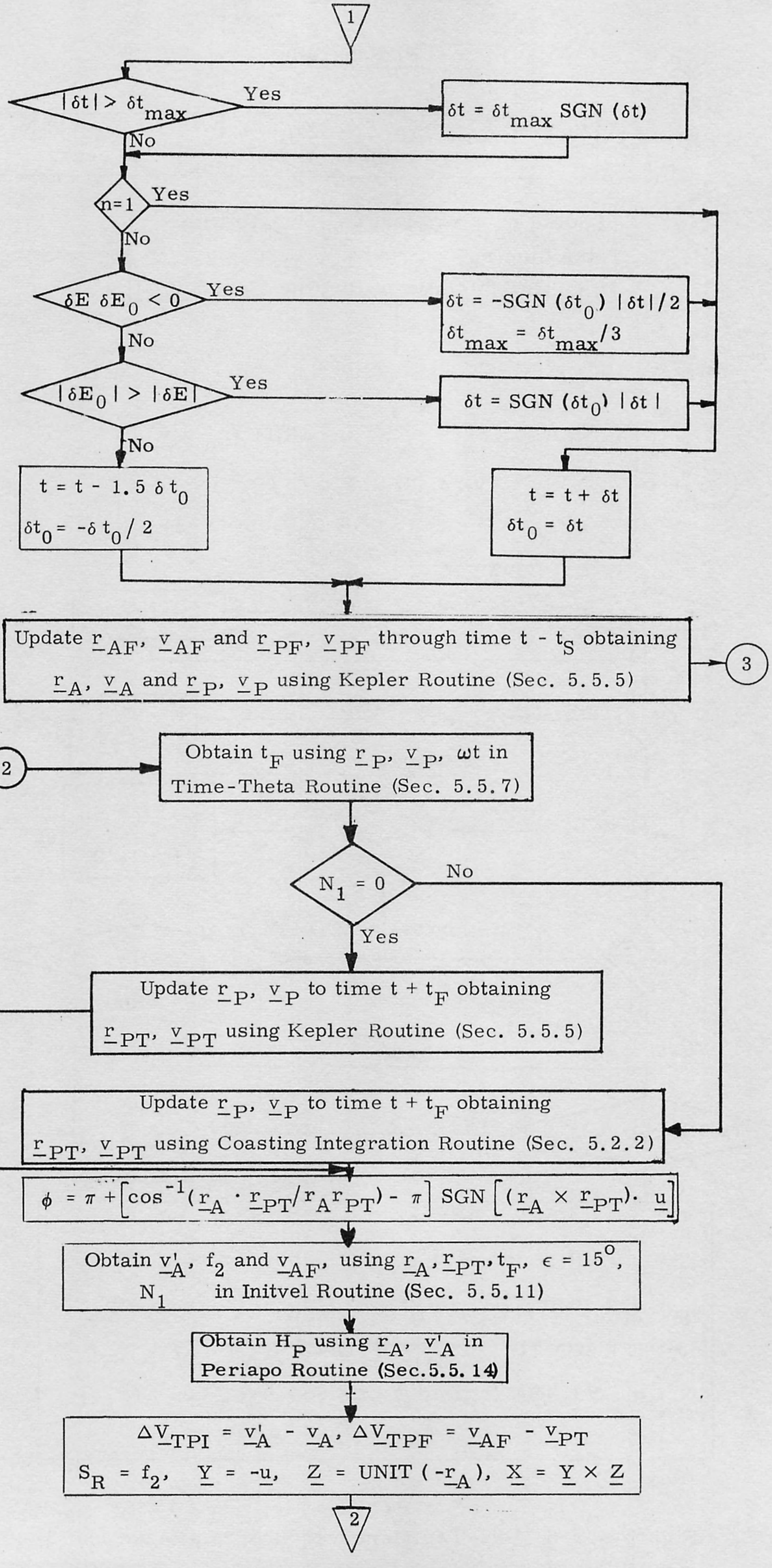


Figure 4.2-5 Pre-TPI Maneuver Program (page 2 of 3)

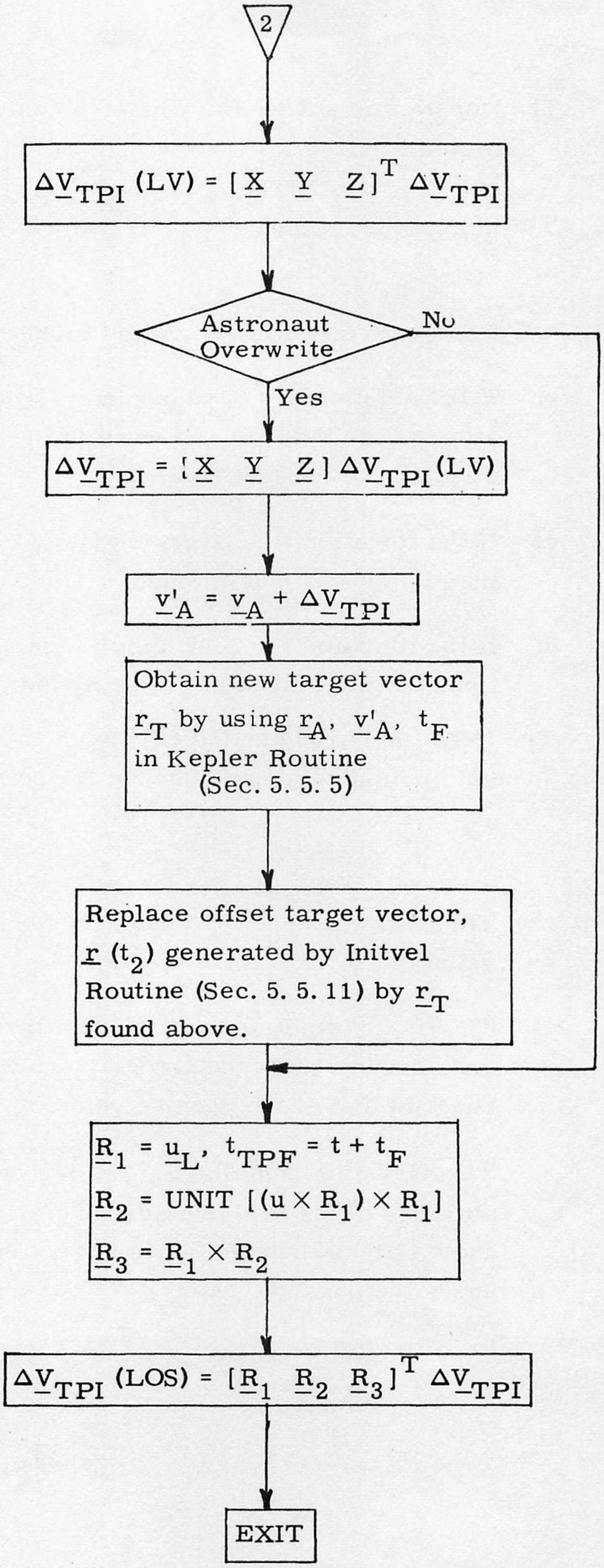


Figure 4. 2-5 **Pre-TPI Maneuver Program (page 3 of 3)**

The iteration is successful when the computed elevation angle E_A is sufficiently close to E.

To help insure convergence, the following steps are taken:

a. The step size δt is restricted to 250 secs.

b. If the solution has been passed $(\delta E \ \delta E_0 < 0)$ the step size is halved and forced in the opposite direction of the last step.

c. If the iteration is converging $(|\delta E_0| - |\delta E| > 0)$, the sign of δt is maintained.

d. If the iteration is proceeding in the wrong direction, the step direction is reversed.

The iteration is terminated for any of the following reasons, and a single alarm as described in Section 4 is set:

1. The iteration counter n has exceeded its maximum value of 15.

2. The line of sight emanating from the active vehicle does not intersect the circular orbit with radius equal to that of the passive vehicle.

3. The elevation angle is inconsistent with the relative altitudes of the two vehicles (e. g., if the elevation angle is less than 180° when the active vehicle is above the passive vehicle).

Upon convergence, the state vectors are precision updated to the TPI time.

The TPI-TPF phase of the program starts with the use of the angle ωt in the Time-Theta Subroutine (Section 5.5.7) to determine the corresponding transfer time t_F. The passive vehicle is next updated through t_F using either a conic or precision updating routine, depending on the value of N_1. The Initvel Routine (Section 5.5.11) is then called to compute a transfer trajectory between the TPI and TPF points with the cone angle ϵ set equal to 15°. The rotation projection switch f_2 is obtained from the Initvel Routine for use in the powered flight steering program as S_R.

The central angle Φ traversed by the active vehicle from the TPI time to intercept is computed as shown in Fig. 4.2-5 for display if requested by the astronaut. This display is used to avoid 180° transfer angle problems. Also, the $\Delta \underline{V}$ in Line-of-Sight coordinates [$\Delta \underline{V}_{TPI}$ (LOS)] is computed as shown in Figure 4.2-5 for display if requested by the crew.

The displays for the Pre-TPI Program are:

1. ΔV_{TPI}

2. ΔV_{TPF}

3. $\Delta \underline{V}_{TPI}(LV)$: Local vertical coordinates

4. Perigee altitude (H_P) following the TPI maneuver

The display of $\Delta \underline{V}_{TPI}(LV)$ may be overwritten by the astronaut. If so, a new target vector is generated as shown in Fig. 4.2-5. This new aimpoint is used for powered flight steering.

In addition, M, MGA and TFI are displayed as in the Pre-CSI program displays.

5.4.2.5 Rendezvous Midcourse Maneuver

This program, corresponding to Program P35 (LM active) or P75 (CSM active) of Section 4, computes a midcourse correction maneuver. This maneuver insures that the active vehicle will intercept the passive vehicle at the time established in the previous Pre-TPI program. The astronaut may call this program any time after the TPI maneuver, but in general no later than 10 minutes before the intercept time. The flow diagram for this program is shown in Fig. 4.2-6.

There is one astronaut input: choice of the active vehicle (P35 LM, P75 CSM). The intercept time t_{TPF} and the number N_1 of precision offsets, both available from the Pre-TPI program, the active $\underline{r}_A$, $\underline{v}_A$, t_A and passive $\underline{r}_P$, $\underline{v}_P$, t_P vehicle state vectors and a time delay are all available in the guidance computer. The time delay ($\delta\tau_3$ for P35 and $\delta\tau_7$ for P75) is the time required to prepare for the thrust maneuver and is stored in either of two erasable locations.

When the program is initiated, the number of navigation measurements since the last maneuver is displayed. In addition, the time since the last maneuver or last maneuver computation is also displayed. Based on this information and additional displays discussed in Section 5.6.7, the astronaut may elect to proceed with the midcourse maneuver at some time. When he does so, the program updates the state vectors to the present time plus the time delay using the Coasting Integration Routine. After updating the passive vehicle to the time t_{TPF} to obtain the target vector $\underline{r}_{PT}$, the Initvel Routine is then called, with the cone angle ϵ set equal to 15°, to obtain the velocity vector on the transfer ellipse. The rotation projection switch f_2 is obtained from the Initvel Routine for use in the powered flight steering program as S_R.

The central angle φ traversed by the active vehicle from the maneuver time to intercept is computed as shown in Fig. 4.2-6 for display if requested by the astronaut. This display is used to avoid 180° transfer angle problems.

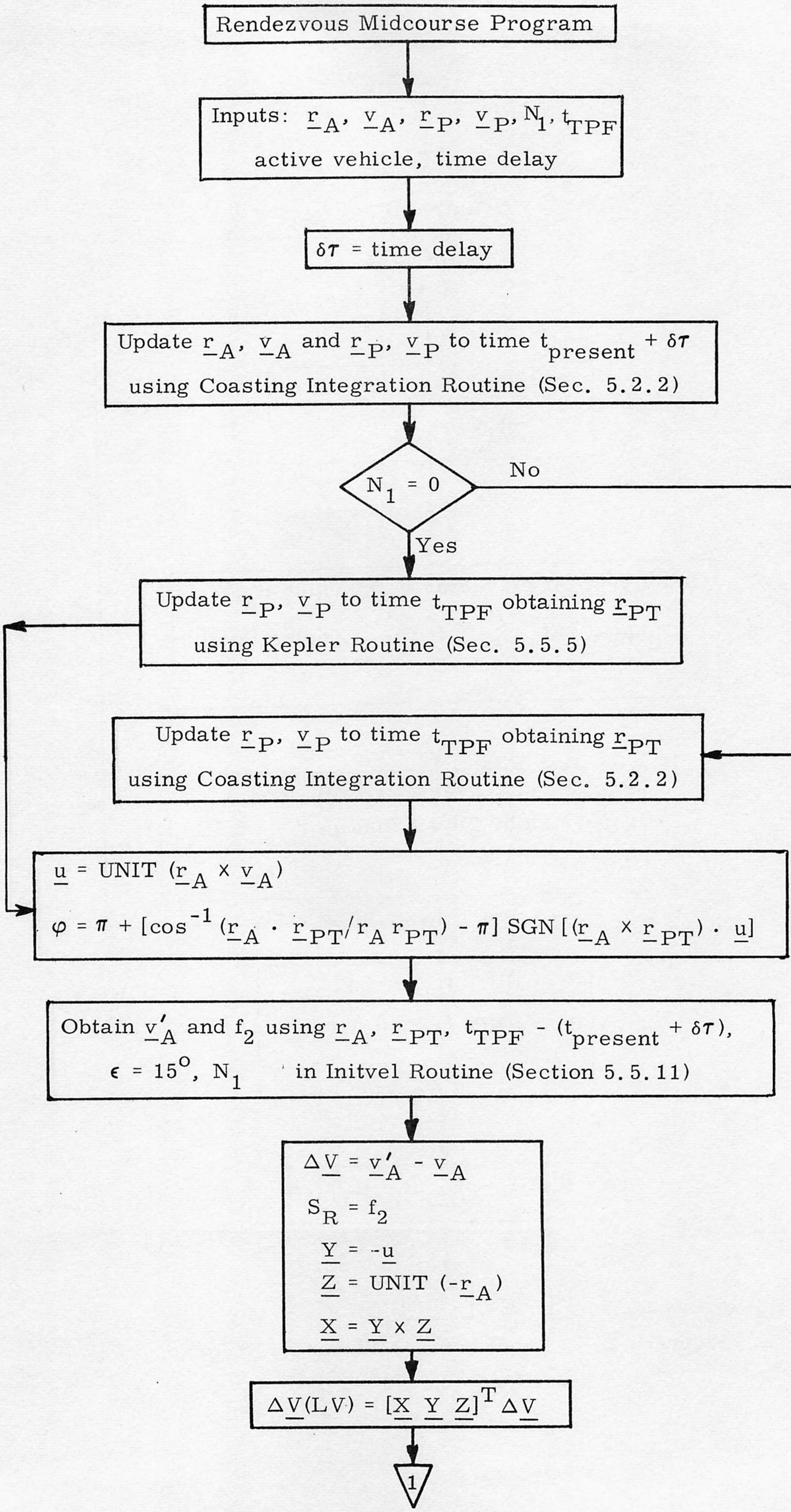


Figure 4.2-6 Rendezvous Midcourse Program (page 1 of 2)

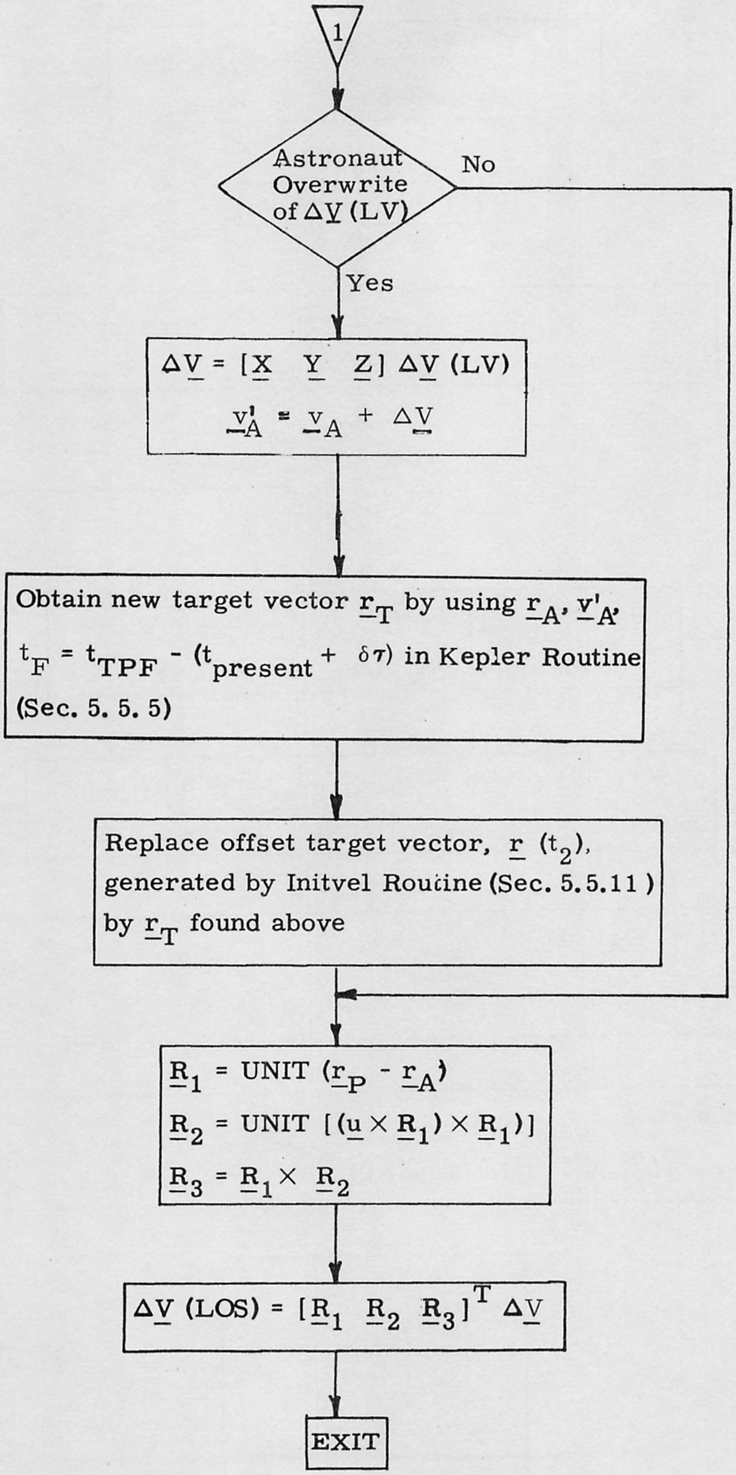


Figure 4.2-6 **Rendezvous** Midcourse **Program**
(page 2 of 2)

After obtaining the maneuver $\Delta \underline{V}$, it is rotated into a local vertical (LV) and a line of sight (LOS) coordinate system obtaining $\Delta \underline{V}$ (LV) which is displayed, and $\Delta \underline{V}$ (LOS) which may be displayed at the request of the crew.

The display of $\Delta \underline{V}$ (LV) may be overwritten by the astronaut. If so, a new target vector is generated as shown in Fig. 4.2-6. This new aimpoint is used for powered flight steering.

In addition, M, MGA and TFI are displayed as in the Pre-CSI program displays.

5.4.3 ABORT TARGETING

5.4.3.1 Aborts From Powered Landing

5.4.3.1.1 General Objectives

The objective of the DPS and APS Abort Programs, P70 and P71 respectively, is to control the abort maneuver such that a suitable injection trajectory is achieved from which a rendezvous profile can be accomplished.

Four of the five target conditions are fixed, as in the ascent. However, the desired down-range speed, $\dot{Z}_D$, is computed as a function of the LM-CSM phase angle and the cutoff altitude. If the phase angle of abort initiation is too large ($\theta > \theta_C$), the LM requires an extra revolution to make the rendezvous. In the P70/P71 initialization, this test is made and the appropriate solution to the phasing problem chosen.

For the cutoff conditions, θ is approximated by the current value, and the perigee radius (R_P) used is an approximate integration of the radial thrust component equation. For an initial value, the nominal cutoff radius (R_D) is used.

The initialization operations are shown in Fig. 4.3-1. The iterative guidance computations are shown in Fig. 3.5-3 along the path followed for P7071FLG=1. This flag is set at the beginning of P70 and P71 to cause the guidance equations to operate in the abort target mode.

The cross range translation to be achieved by the abort maneuver is automatically limited by the P70/P71 target initialization to a pad-loaded maximum (1/2 deg, approximately 8 nm, was used for Apollo 14).

The DPS Abort Program, P70, is called by the LGC upon receipt of an abort discrete bit, while P71 is called automatically when an abort stage discrete bit is sensed by the LGC. However, if the astronaut has set CHANBKUP bit 1 to 1, both the abort and abort stage discretes will be ignored by the LGC. Either program may be called by the astronaut using a V37 DSKY entry, regardless of the state of CHANBKUP bit 1.

When P70 or P71 is called to supersede any of the landing maneuver programs (P63, P64, and P66) their initialization and operation are very similar, both using the Powered Ascent Guidance Equations to control the vehicle to the desired injection conditions. The input parameters required for the P70 and P71 abort programs when used in this mode, which are supplied by the landing programs are:

1. t_{IG}(Descent) - time of DPS landing maneuver ignition

2. t, $\underline{r}(t)$, $\underline{v}(t)$ - current LM state vector

3. $\underline{r}_c$, $\underline{v}_c$ - any recent CSM state vector

4. m - vehicle mass (used only by P70)

All other input parameters are prestored in the LGC.

The primary outputs of P70 and P71 are the same as those for the Powered Ascent Program, P12, of Section 5.3.5, namely: LM Digital Autopilot (DAP) attitude commands, engine-off signals, and display parameters for the Vertical Rise and Ascent Guidance Phases (Section 5.3.5.3).

5.4.3.1.2 DPS Abort Program, P70

The functional logic for P70 is shown in Fig. 4.3-1, together with the Ascent Guidance Computations, Sec. 5.3.5.9. When a DPS powered abort is initiated, the thrust filter must be initialized. Because of the decrease of vehicle mass during the descent burn preceding the abort, the vehicle dynamics must be related to the current mass of the vehicle. In addition, the engine performance itself varies because of erosion which takes place during burning. The main effect of the erosion is to increase the mass flow rate, $\dot{m}$, leaving the exhaust velocity, V_e, relatively unchanged. The initialization of P70 assumes a constant $\dot{m}$ which introduces an error into the initialization of the engine performance and the thrust magnitude filter. This error will decrease and eventually vanish. The engine performance initialization takes place as follows:

$$a_T = \frac{\dot{m}\, V_e}{m} = \frac{V_e}{\tau} \qquad (4.3.1)$$

where

$$\tau = \frac{m}{\dot{m}}$$

The thrust filter is initialized with the reciprocals of three dummy PIPA readings, ΔV_1, ΔV_2, ΔV_3. The same number is used for all three

$$\frac{1}{\Delta V_1} = \frac{1}{\Delta V_2} = \frac{1}{\Delta V_3} = \frac{1}{\Delta V_{IG}}$$

where

$$\frac{1}{\Delta V_{IG}} = \frac{m_{IG}}{K(1/DV)\ \Delta t} \ ,$$

$K\ (1/DV) = \dot{m} V_e$

and Δt = computation cycle time.

The final engine performance parameter to be initialized is

$\Delta t_{Tail\text{-}off}$ - a negative constant time increment used to correct t_{go} for the P70 DPS tail-off. It is initialized from $\Delta t_{Tail\text{-}off}$ (P70) in fixed memory.

The next step is to set a mission sequence flag, FLP70, to indicate that P70 has been called, so that any subsequent call to P71 will take this fact into account.

The initialization of the target conditions is done next. First, the time-to-go is estimated by

$$t_{go} = TFI$$

where

$$TFI = t - t_{IG} \text{ (Descent)}$$

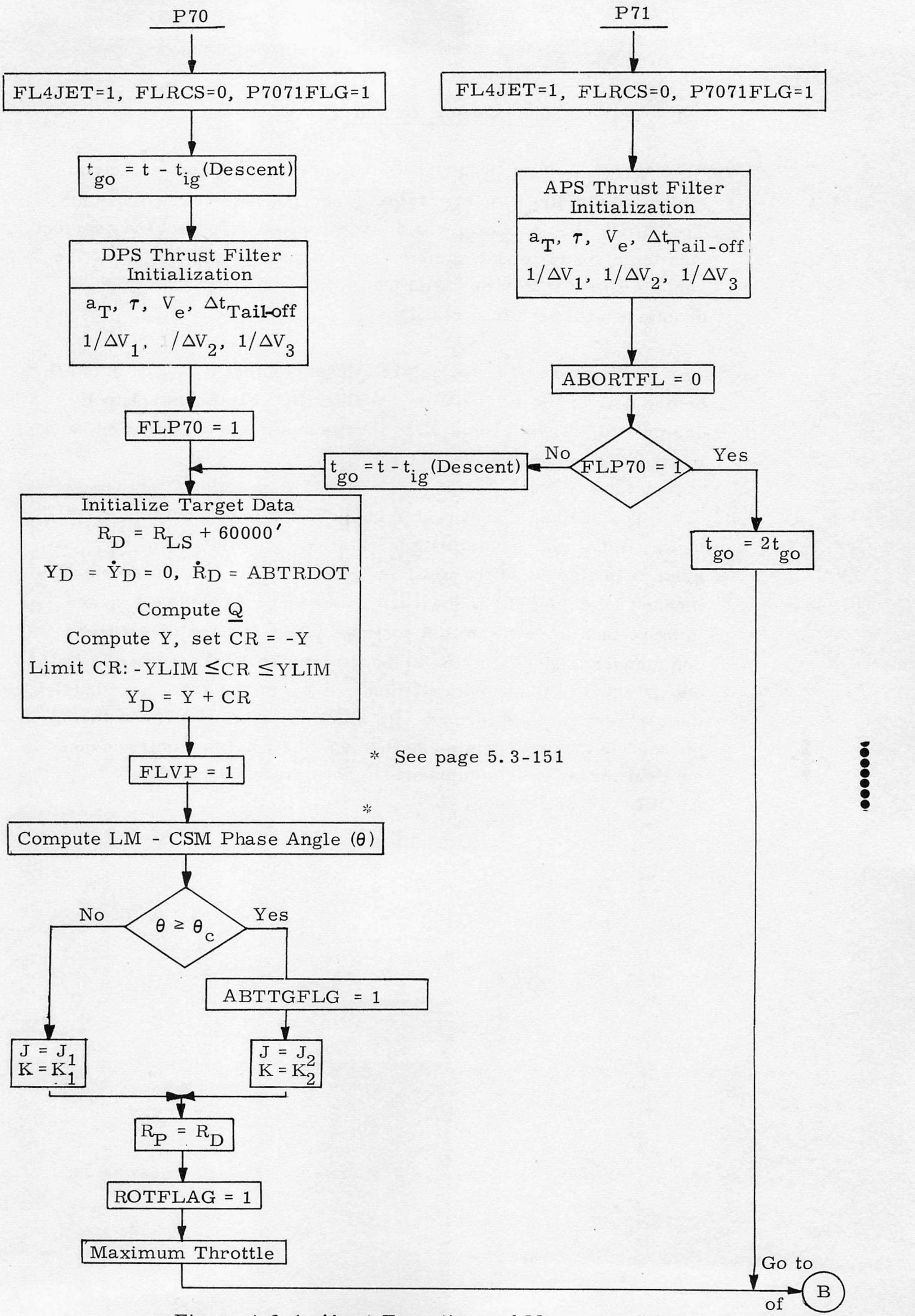
The desired cutoff conditions are transferred from fixed to erasable storage, and $\underline{Q}$, the normal to the CSM orbit plane, is computed.

5.4.3.1.3 APS Abort Program, P71

The APS Abort Program, P71, can be used in two modes, either as a primary abort program, or as a follow-up to P70. In either mode, the thrust filter initialization is the same, and differs from the P70 initialization in that all parameters are prestored in the LGC. The initialization of t_{go} is different in the two modes. In the primary mode it is initialized in the same way as in P70. In the follow-up mode, a value of t_{go} is available from the operation of P70, and this is doubled to account for the lower acceleration of a full ascent stage compared with an empty descent stage. The initialization of the target is bypassed in the follow-up mode, since it has already been done by P70.

When P71 is called, the flag ABORTFL is set to 0 to prohibit any further calls to the abort programs P70 and P71.

ABTTGFLG is used for downlink information to indicate which set of E-load parameters are being used (when set J_2, K_2; when reset J_1, K_1). This flag is reset by R00 in the V37 selection of any program except P70 or P71, and set by the selection of J_2, K_2 within P70 or P71.



* See page 5.3-151

Figure 4.3-1 Abort Targeting and Maneuver Control from Powered Lunar Landing

5.4.3.1.4 Abort Rotation Control

When an abort is initiated, it is necessary to put some constraint on the direction of rotation from initial to desired attitude. The FINDCDUW program would, if left to itself, select the shortest rotation arc and, under some circumstances, this could cause the vehicle x-axis to rotate down through the local vertical, causing problems with certain antennas.

To prevent this, the desired thrust vector, $\underline{u}_{FDP}$, is modified before delivery to FINDCDUW. A flag, ROTFLAG, is set to 1 during P70/P71 initialization to invoke this mode of operation, and it can be followed on page 4 of Fig. 3.5-3.

The desired thrust vector is above the local horizontal; hence, a downward-pointing $\underline{X}_B$ which is within 90 deg of $\underline{u}_{FDP}$ must be posigrade of the local vertical, and $\underline{X}_B$ will not rotate through $-\underline{R}$ during the maneuver. If the angle between $\underline{u}_{FDP}$ and $\underline{X}_B$ is more than θ_1, the rotation control causes vertical thrust to be commanded. The value for θ_1 loaded for APOLLO 14 was 90 deg, consistent with the above criterion. If $\underline{X}_B$ is above the horizontal, the shortest route is through $+\underline{R}$, which is satisfactory. If the angle between $\underline{X}_B$ and $+\underline{R}$ is more than θ_2, the rotation control causes vertical thrust to be commanded. The value of θ_2 loaded for APOLLO 14 was 30 deg.

5.5 BASIC SUBROUTINES

5.5.1 GENERAL COMMENTS

The basic solar system and conic trajectory subroutines which are used by the various guidance and navigation routines are described in this section.

5.5.1.1 Solar System Subroutines

The subroutines used to determine the translation and rotation of the relevant solar system bodies (earth, moon and sun) are designed specifically for a fourteen day lunar landing mission. The method of computing the moon and the sun lines - of - sight relative to the earth is given in Section 5.5.4. The transformations between the Basic Reference Coordinate System and the Earth - and Moon-fixed Coordinate Systems are described in Section 5.5.2. The procedure for transforming between vectors in the Basic Reference Coordinate System and latitude, longitude, altitude coordinates is given in Section 5.5.3. Although these subroutines are normally used in the lunar landing mission, they are valid for use in any mission of not more than fourteen days duration in earth-moon space.

5.5.1.2 Conic Trajectory Subroutines

This is a description of a group of conic trajectory subroutines which are frequently used by higher level routines and programs in both the Command Module and the Lunar Module computers.

Those subroutines, whose block diagrams are presented in Sections 5.5.5 to 5.5.10, provide solutions to the following conic problems. (See nomenclature which follows)

(1) Given $\underline{r}(t_1)$, $\underline{v}(t_1)$, t_D; solve for $\underline{r}(t_2)$, $\underline{v}(t_2)$

(Kepler Subroutine)

(2) Given $\underline{r}(t_1)$, $\underline{r}(t_2)$, t_{D21}, s_G; solve for $\underline{v}(t_1)$

(Lambert Subroutine)

(3) Given $\underline{r}(t_1)$, $\underline{v}(t_1)$, θ ; solve for t_{21}, $\underline{r}(t_2)$, $\underline{v}(t_2)$

(Time-Theta Subroutine)

(4) Given $\underline{r}(t_1)$, $\underline{v}(t_1)$, $r(t_2)$, s_r; solve for t_{21}, $\underline{r}(t_2)$, $\underline{v}(t_2)$

(Time-Radius Subroutine)

(5) Given $\underline{r}(t)$, $\underline{v}(t)$; solve for r_P, r_A, e

(Apsides Subroutine)

In addition, the following useful subroutines are provided.

(6) Conic Parameters Subroutine (See Fig. 5.10-1).

(7) Geometric Parameters Subroutine (See Fig. 5.10-2).

(8) Iterator Subroutine (See Fig. 5.10-3).

The solutions to the above set of conic problems have stringent accuracy requirements. Programming the fixed-point Apollo computer introduces two constraints which determine accuracy limitations: the 28 bit double precision word length, and the range of variables which is several orders of magnitude for the Apollo Mission.

In order to maintain numerical accuracy when these subroutines are programmed into the Apollo Computer, floating point programming techniques must be exercised. The effect is for even a simple equation to require a large number of computer instructions. The alternative to this is to separate the problem into phases, each with a different variable range. This, however, requires an even larger number of instructions. These considerations provide the incentive for efficiently organizing the conic equations as shown on the block diagrams.

In addition to the requirement for accuracy the solution to the Kepler and Lambert Problems must be accomplished in a minimum of computation time in order that the guidance system operate satisfactorily in real time. This additional constraint dictates that a minimum of computer instructions be performed when solving the problem.

Method of Solution

To minimize the total number of computer instructions, the problems are solved in the "universal" form; i.e. only equations which are equally valid for the ellipse, parabola and hyperbola are used. Also these subroutines can be used with either the earth or the moon as the attracting body.

Kepler's equation, in the universal form, is utilized to relate transfer time to the conic parameters. All other necessary equations are also universal. The Kepler and Lambert problems are solved with a single iteration loop utilizing a simple first-order slope iterator. In the case of the Kepler problem a third order approximation is available to produce the initial guess for the independent variable (See Eq. (2.2.4) of Section 5.2.2.2).

Sections 5.5.5 thru 5.5.10 provide block diagrams of the detailed computational procedures for solving the various problems. The equations are presented in block diagram form with nomenclature below.

Range of Variables

As indicated previously, the programming of the conic subroutines requires a careful balance between accuracy, computational speed and number of instructions. This balance, in the Apollo Guidance Computer, leaves very little margin in any of these areas.

Since the values of problem variables are determined by the solution of the problem being solved and since the problem may originate from the ground system, it is essential that the variable range limitations be defined. The conic routines are incapable of handling problems when the solution lies outside of the range.

The following is a list of the maximum allowable numeric values of the variables. Note that, in addition to fundamental quantities such as position and velocity, there are limitations on intermediate variables and combinations of variables.

Scaling for Conic Subroutines (Sections 5.5.5 to 5.5.10)

Parameter	Maximum Value* Earth Primary Body	Maximum Value* Moon Primary Body
r	2^{29}	2^{27}
v	2^{7}	2^{5}
t	2^{28}	2^{28}
α**	2^{-22}	2^{-20}
α_N**	2^{6}	2^{6}
p_N	2^{4}	2^{4}
$\cot \gamma$	2^{5}	2^{5}
$\cot \frac{\theta}{2}$	2^{5}	2^{5}
x	2^{17}	2^{16}
$\xi = \alpha x^2$***	-50	-50
	$+4\pi^2$	$+4\pi^2$
$c_1 = \underline{r} \cdot \underline{v}/\sqrt{\mu}$	2^{17}	2^{16}
$c_2 = r v^2/\mu - 1$	2^{6}	2^{6}
$\lambda = r(t_1)/r(t_2)$	2^{7}	2^{7}
$\cos \theta - \lambda$	2^{7}	2^{7}

* All dimensional values are in units of meters and centiseconds.

** The maximum absolute value occurs for negative values of this parameter.

***Both the maximum and minimum values are listed since neither may be exceeded.

Parameter	Maximum Value* Earth	Moon
e	2^3	2^3
x^2	2^{34}	2^{32}
$x^2\ c(\xi)$	2^{33}	2^{31}
$x^3\ s(\xi)/\sqrt{\mu}$	2^{28}	2^{28}
$c_1\ x^2\ c(\xi)$	2^{49}	2^{46}
$c_2\ x^2\ s(\xi)$	2^{35}	2^{33}
$x\ [c_2\ x^2\ s(\xi) + r(t_1)]$	2^{49}	2^{46}
$\xi\ s(\xi)$	2^7	2^7
$x^2\ c(\xi)/r$	2^8	2^8
$\sqrt{\mu}\ x\ (\xi\ s(\xi) - 1)/r(t_2)$	2^{15}	2^{13}
$c(\xi)$	2^4	2^4
$s(\xi)$	2^1	2^1

* All dimensional values are in units of meters and centiseconds.

Nomenclature for Conic Subroutines (Sections 5.5.5 to 5.5.10)

$\underline{r}(t_1)$	initial position vector
$\underline{v}(t_1)$	initial velocity vector
$\underline{r}(t_2)$	terminal position vector
$\underline{v}(t_2)$	terminal velocity vector
$\underline{u}_N$	unit normal in the direction of the angular momentum vector
α	reciprocal of semi-major axis (negative for hyperbolas)
r_P	radius of pericenter
r_A	radius of apocenter
e	eccentricity
α_N	ratio of magnitude of initial position vector to semi-major axis
p_N	ratio of semi-latus rectum to initial position vector magnitude
γ	inertial flight path angle as measured from vertical
θ	true anomaly difference between $\underline{r}(t_1)$ and $\underline{r}(t_2)$
f	true anomaly of r (t_2)

x	a universal conic parameter equal to the ratio of eccentric anomaly difference to $\sqrt{+\alpha}$ for the ellipse, or the ratio of the hyperbolic analog of eccentric anomaly difference to $\sqrt{-\alpha}$ for the hyperbola
x'	value of x from the previous Kepler solution
t_{21}	computed transfer time from Kepler's equation $(t_2 - t_1)$
t'_{21}	transfer time corresponding to the previous solution of Kepler's equation
t_D	desired transfer time through which the conic update of the state vector is to be made
t_{D21}	desired transfer time to traverse from $\underline{r}(t_1)$ to $\underline{r}(t_2)$
t_{ERR}	error in transfer time
ϵ_t	fraction of desired transfer time to which t_{ERR} must converge
Δx	increment in x which will produce a smaller t_{ERR}
ϵ_x	value of Δx which will produce no significant change in t_{21}
$\Delta \cot \gamma$	increment in $\cot \gamma$ which will decrease the magnitude of t_{ERR}
ϵ_c	value of $\Delta \cot \gamma$ which will produce no significant change in t_{21}

μ	product of universal gravitational constant and mass of the primary attracting body
x_{MAX}	maximum value of x
x_{MIN}	minimum value of x
cot_{MAX}	maximum value of cot γ
cot_{MIN}	minimum value of cot γ
ℓ_{MAX}	upper bound of general independent variable
ℓ_{MIN}	lower bound of general independent variable
x_{MAX1}	absolute upper bound on x with respect to the moon
x_{MAX0}	absolute upper bound on x with respect to the earth
k	a fraction of the full range of the independent variable which determines the increment of the independent variable on the first pass through the iterator
y	general dependent variable
y'	previous value of y
y_{ERR}	error in y
z	general independent variable

Δz	increment in z which will produce a smaller y_{ERR}
s_G	a sign which is plus or minus according to whether the true anomaly difference between $\underline{r}(t_1)$ and $\underline{r}(t_2)$ is to be less than or greater than 180 degrees
$s_{\dot{r}}$	a sign which is plus or minus according to whether the desired radial velocity at $\mathbf{r}(t_2)$ is plus or minus
$\underline{\eta}_1$	general vector # 1
$\underline{\eta}_2$	general vector # 2
ϕ	angle between $\underline{\eta}_1$ and $\underline{\eta}_2$
f_1	a switch set to 0 or 1 according to whether a guess of cot γ is available or not
f_2	a switch set to 0 or 1 according to whether Lambert should determine $\underline{u}_N$ from $\underline{r}(t_1)$ and $\underline{r}(t_2)$ or $\underline{u}_N$ is an input
f_3	a tag set to 0 or 1 according to whether the iterator should use the "Regula Falsi" or bias method
f_4	a flag set to 0 or 1 according to whether the iterator is to act as a first order or a second order iterator
f_5	a flag set to 0 or 1 according to whether Lambert converges to a solution or not

f_6	a switch set to 0 or 1 according to whether or not the new state vector is to be an additional output requirement of the Time-Theta or Time-Radius problems.
f_7	a flag set to 1 if the inputs require that the conic trajectory must close through infinity
f_8	a flag set to 1 if the Time-Radius problem was solved for pericenter or apocenter instead of $r(t_2)$
f_9	a flag set to 1 if the input to the Time-Radius Subroutine produced an e less than 2^{-18}
t_p	period of the orbit
k_1	the minimal acceptance fraction of t_{D21} to which t_{ERR} must converge
n_1	a flag set to 0 or 1 according to whether or not the velocity vector at the terminal position is to be an additional output requirement of the Lambert Routine

5.5.2 PLANETARY INERTIAL ORIENTATION SUBROUTINE

This subroutine is used to transform vectors between the Basic Reference Coordinate System and a Planetary (Earth-fixed or Moon-fixed) Coordinate System at a specified time. These three coordinate systems are defined in Section 5.1.4.

Let $\underline{r}$ be a vector in the Basic Reference Coordinate System, $\underline{r}_P$ the same vector expressed in the Planetary Coordinate System, and t the specified ground elapsed time (GET). Then,

$$\underline{r}_P = M(t)\,(\underline{r} - \underline{\ell} \times \underline{r}) \qquad (5.2.1)$$

and

$$\underline{r} = M^T(t)\,(\underline{r}_P + \underline{\ell}_P \times \underline{r}_P) \qquad (5.2.2)$$

where M(t) is a time dependent orthogonal transformation matrix, $\underline{\ell}$ is a small rotation vector in the Basic Reference Coordinate System, and $\underline{\ell}_P$ is the same vector $\underline{\ell}$ expressed in the Planetary Coordinate System. The vector $\underline{\ell}$ is considered constant in one coordinate system for the duration of the mission. The method of computing M(t) and $\underline{\ell}$ depends on whether the relevant planet is the earth or the moon.

Case I - Earth

For the earth, the matrix M(t) describes a rotation about the polar axis of the earth (the Z-axis of the Earth-fixed Coordinate

System), and the vector $\underline{\ell}$ accounts for the precession and nutation of the polar axis (the deviation of the true pole from the mean pole).

Let A_X and A_Y be the small angles about the X- and Y-axes of the Basic Reference Coordinate System, respectively, that describe the precession and nutation of the earth's polar axis. The values of these two angles at the midpoint of the mission are included in the pre-launch erasable data load and are considered constant throughout the flight. Then,

$$\underline{\ell} = \begin{pmatrix} A_X \\ A_Y \\ 0 \end{pmatrix}$$

$$A_Z = A_{Z0} + \omega_E (t + t_0) \qquad (5.2.3)$$

$$M(t) = \begin{pmatrix} \cos A_Z & \sin A_Z & 0 \\ -\sin A_Z & \cos A_Z & 0 \\ 0 & 0 & 1 \end{pmatrix}$$

$$\underline{\ell}_P = M(t)\, \underline{\ell}$$

where A_{Z0} is the angle between the X-axis of the Basic Reference Coordinate System and the X-axis of the Earth-fixed Coordinate System (the intersection of the Greenwich meridian and the equatorial plane of the earth) at July 1.0, 1971 universal time (i.e., midnight at Greenwich just prior to July 1, 1971), t_0 is the elapsed time between July 1.0, 1971 universal time and the time that the computer clock was zeroed, and ω_E is the angular velocity of the earth.

Case II - Moon

For the moon, the matrix M(t) accounts for the difference in orientation of the Basic Reference and Moon-fixed Coordinate Systems in exact accordance with Cassini's laws, and the rotation vector $\underline{\ell}$ corrects for deviations from the above orientation because of physical libration.

Define the following three angles which are functions of time:

B = the obliquity, the angle between the mean earth equatorial plane and the plane of the ecliptic.

Ω_I = the longitude of the node of the moon's orbit measured from the X-axis of the Basic Reference Coordinate System.

F = the angle from the mean ascending node of the moon's orbit to the mean moon.

Let I be the constant angle between the mean lunar equatorial plane and the plane of the ecliptic (5521.5"). Then, the sequence of rotations which brings the Basic Reference Coordinate System into coincidence with the Moon-fixed Coordinate System (neglecting libration) is as follows:

Rotation	Axis of Rotation	Angle of Rotation
1	X	B
2	Z	Ω_I
3	X	-I
4	Z	π + F

The transformation matrices for these rotations are, respectively,

$$M_1 = \begin{pmatrix} 1 & 0 & 0 \\ 0 & \cos B & \sin B \\ 0 & -\sin B & \cos B \end{pmatrix}$$

$$M_2 = \begin{pmatrix} \cos \Omega_I & \sin \Omega_I & 0 \\ -\sin \Omega_I & \cos \Omega_I & 0 \\ 0 & 0 & 1 \end{pmatrix}$$

$$M_3 = \begin{pmatrix} 1 & 0 & 0 \\ 0 & \cos I & -\sin I \\ 0 & \sin I & \cos I \end{pmatrix} \qquad (5.2.4)$$

$$M_4 = \begin{pmatrix} -\cos F & -\sin F & 0 \\ \sin F & -\cos F & 0 \\ 0 & 0 & 1 \end{pmatrix}$$

The matrix M(t) is then given by

$$M(t) = M_4 \, M_3 \, M_2 \, M_1 \qquad (5.2.5)$$

The following approximate method is used to determine the transformation between the Basic Reference and Moon-fixed Coordinate Systems.

The angles B, Ω_I and F are computed as linear functions of time. Let $\underline{\ell}_M$ be the value of the vector libration $\underline{\ell}_P$ (expressed in the Moon-fixed Coordinate System) at the nominal lunar landing time. While the LM is on the surface of the moon, a new value of $\underline{\ell}_M$ referenced to the nominal launch time should be uplinked. The vector $\underline{\ell}_M$ is included in the pre-launch erasable data load and is considered constant throughout the flight. Then,

$$\underline{\ell}_P = \underline{\ell}_M$$

$$t_M = t + t_0$$

$$B = B_0 + \dot{B}\, t_M$$

$$\Omega_I = \Omega_{I0} + \dot{\Omega}_I \, t_M$$

$$F = F_0 + \dot{F}\, t_M$$

$$\underline{a} = \begin{pmatrix} \cos \Omega_I \\ \cos B \sin \Omega_I \\ \sin B \sin \Omega_I \end{pmatrix} \qquad (5.2.6)$$

$$\underline{b} = \begin{pmatrix} -\sin \Omega_I \\ \cos B \cos \Omega_I \\ \sin B \cos \Omega_I \end{pmatrix}$$

$$\underline{c} = \begin{pmatrix} 0 \\ -\sin B \\ \cos B \end{pmatrix}$$

$$\underline{d} = \underline{b}\, C_I - \underline{c}\, S_I$$

$$\underline{m}_2 = \underline{b}\, S_I + \underline{c}\, C_I$$

$$\underline{m}_0 = -\underline{a} \cos F - \underline{d} \sin F \qquad \text{(5.2.6) (cont.)}$$

$$\underline{m}_1 = \underline{a} \sin F - \underline{d} \cos F$$

$$M(t) = \begin{pmatrix} \underline{m}_0^T \\ \underline{m}_1^T \\ \underline{m}_2^T \end{pmatrix}$$

$$\underline{\ell} = M^T(t) \cdot \underline{\ell}_P$$

where B_0, Ω_{I0}, and F_0 are the values of the angles B, Ω_I and F, respectively, at July 1.0, 1971 universal time; $\dot{B}$, $\dot{\Omega}_I$ and $\dot{F}$ are the rates of change of these angles; and C_I and S_I are the cosine and sine, respectively, of the angle I.

5.5.3 LATITUDE-LONGITUDE SUBROUTINE

For display and data load purposes, the latitude, longitude, and altitude of a point near the surface of the earth or the moon are more meaningful and more convenient to use than the components of a position vector. This subroutine is used to transform position vectors between the Basic Reference Coordinate System and Geographic or Selenographic latitude, longitude, altitude at a specified time.

In the case of the moon, the altitude is computed above either the landing site radius, r_{LS}, or the mean lunar radius, r_M. For the earth, the altitude is defined with respect to either the launch pad radius, r_{LP}, or the radius of the Fischer ellipsoid, r_F, which is computed from

$$r_F^2 = \frac{b^2}{1 - (1 - \frac{b^2}{a^2})(1 - \mathrm{SINL}^2)} \qquad (5.3.1)$$

where a and b are the semi-major and semi-minor axes of the Fischer ellipsoid, respectively, and SINL is the sine of the geocentric latitude.

The computational procedures are illustrated in Figs. 5.3-1, 5.3-2, and 5.3-3. The calling program must specify either a vector $\underline{r}$ or latitude (Lat), longitude (Long), and altitude (Alt). In addition, the program must set the time t and the two indicators P and F where

$$P = \begin{cases} 0 \text{ for earth} \\ 1 \text{ for moon} \end{cases}$$

$$F = \begin{cases} 1 & \text{for Fischer ellipsoid or mean lunar radius} \\ 0 & \text{for launch pad or landing site radius} \end{cases}$$

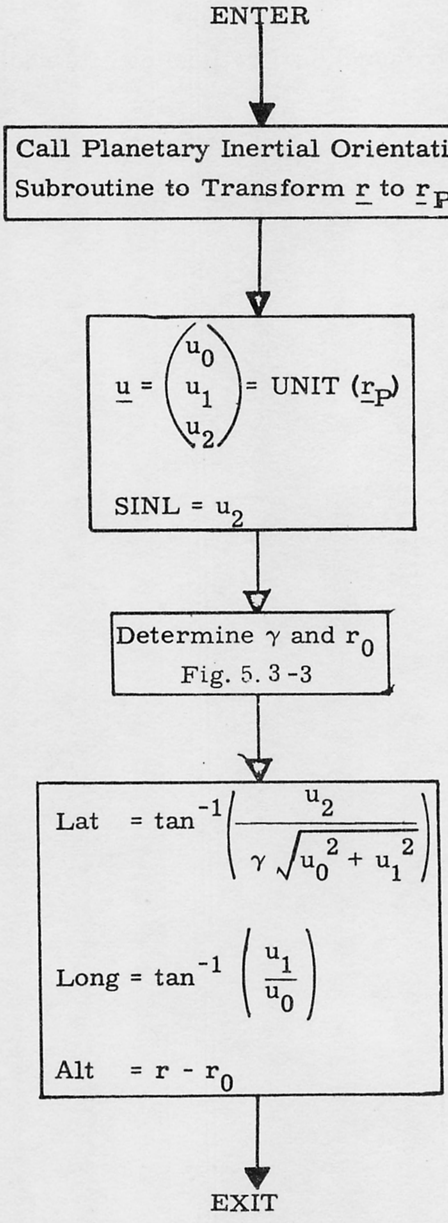


Fig. 5.3-1 Vector to Latitude, Longitude, Altitude Computation Logic Diagram

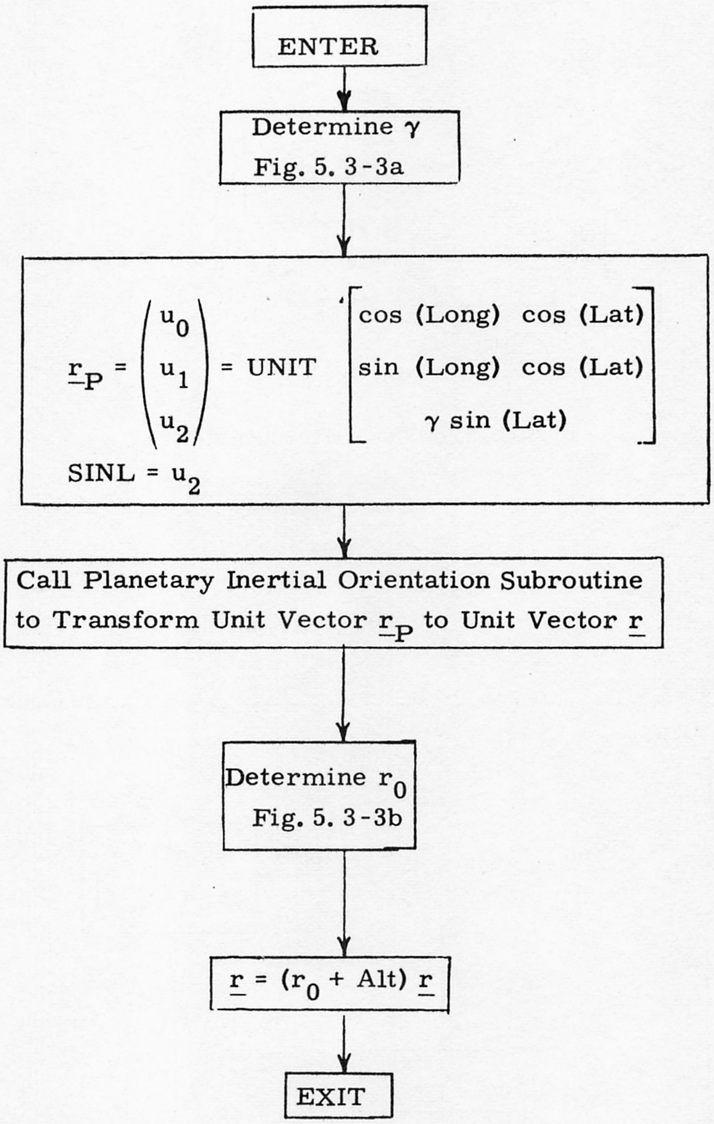


Fig. 5. 3-2 Latitude, Longitude, Altitude to Vector Computation Logic Diagram

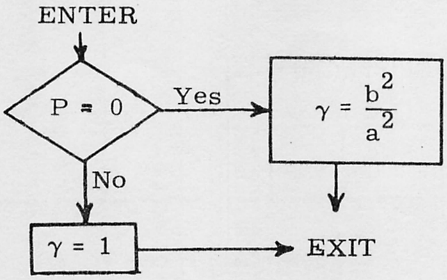


Figure 5. 3-3a Determination of γ

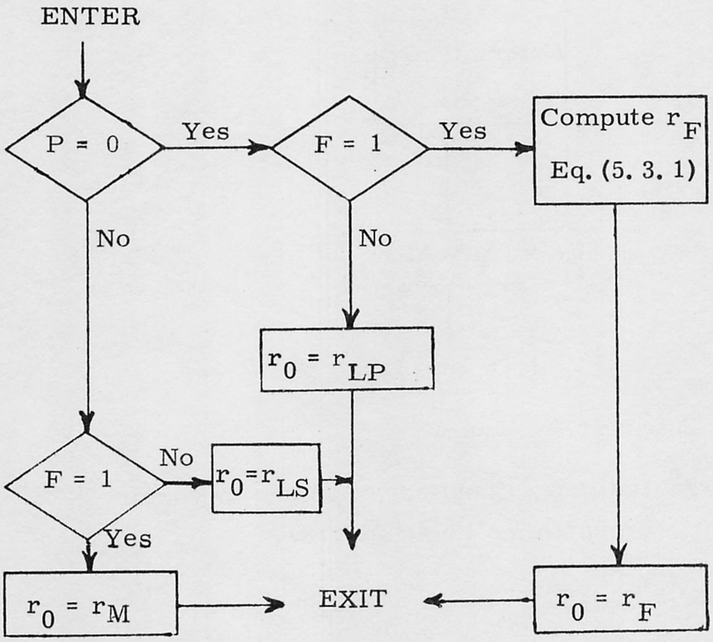


Figure 5. 3-3b Determination of r_0

5.5.4 LUNAR AND SOLAR EPHEMERIDES

This subroutine is called by LOCSAM(Section 5.5.13) in order to obtain the line-of-sight unit vectors of the moon and sun, with respect to the LM, which may be used for IMU alignment.

The positions of the moon and sun are obtained from simple approximation schemes which are stored in the computer's fixed memory.

If the obliquity of the ecliptic and the inclination of the lunar orbit are considered constant, the unit position vector of the moon with respect to the earth in the Basic Reference Coordinate System defined in Section (5.1.4), is given by

$$\underline{u}_{EM} = \text{UNIT}\begin{bmatrix} \cos(LOM) \\ K_1 \sin(LOM) - K_2 \sin(LOM - LON) \\ K_3 \sin(LOM) + K_4 \sin(LOM - LON) \end{bmatrix}$$

where the longitude of the mean moon (LOM) is approximated using its value and rate at the appropriate July 1.0, 1971, and using empirically determined periodic terms:

$$LOM = LOM_O + LOM_R t - \Big[A \sin(OMEGA_A t + PHASE_A) + B \sin(OMEGA_B t + PHASE_B)\Big]$$

and where the longitude of the moon's node (LON) is similarly approximated:

5.5.5 KEPLER SUBROUTINE

The Kepler Subroutine solves for the two body position and velocity vectors at the terminal position given the initial position and velocity vectors and a transfer time to the terminal position.

This section contains information to aid the reader in understanding the less obvious aspects of the Kepler Subroutine block diagram depicted in Figs. 5.5-1 thru 5.5-3. The subroutines referred to in these figures are presented in Section 5.5.10 and the nomenclature is found in Section 5.5.1.2.

Prior to entering the Kepler Subroutine, an initial estimate of x can be generated via Eq. (2.2.4) of Section 5.2.2.2 with $\frac{\Delta t}{2} = t_D - t_{21}$ and $\tau = t_D$. However, x' and t_{21} are non-zero only if the subroutine is being used repetitively.

Although, theoretically, there is no upper bound on x, the practical bound is set to x_{MAX0} or x_{MAX1} to eliminate non-feasible trajectories and increase the accuracy to which x can be computed. In addition, αx^2 has a practical range of $-50 < \alpha x^2 < (2\pi)^2$ which determines an independent upper bound on x. The x_{MAX} used, then, corresponds to the smaller of the two values.

The transfer time convergence criterion is approximately the same as the granularity of the time input. Since, for some of the problems to be solved, the sensitivity of time to x is so large that the granularity in x, ϵ_x, produces a change in time which exceeds the granularity in time, it is necessary to introduce ϵ_x as a redundant convergence criterion.

The Kepler Subroutine, provided the parameter range constraints are satisfied, will always produce a solution.

A negative value of t_D will cause the subroutine to update the state vector backward in time (i. e. backdate the state vector). The subroutine may be called to update or backdate for any amount of time; there are no restrictions on whether the time t_D is less than a period.

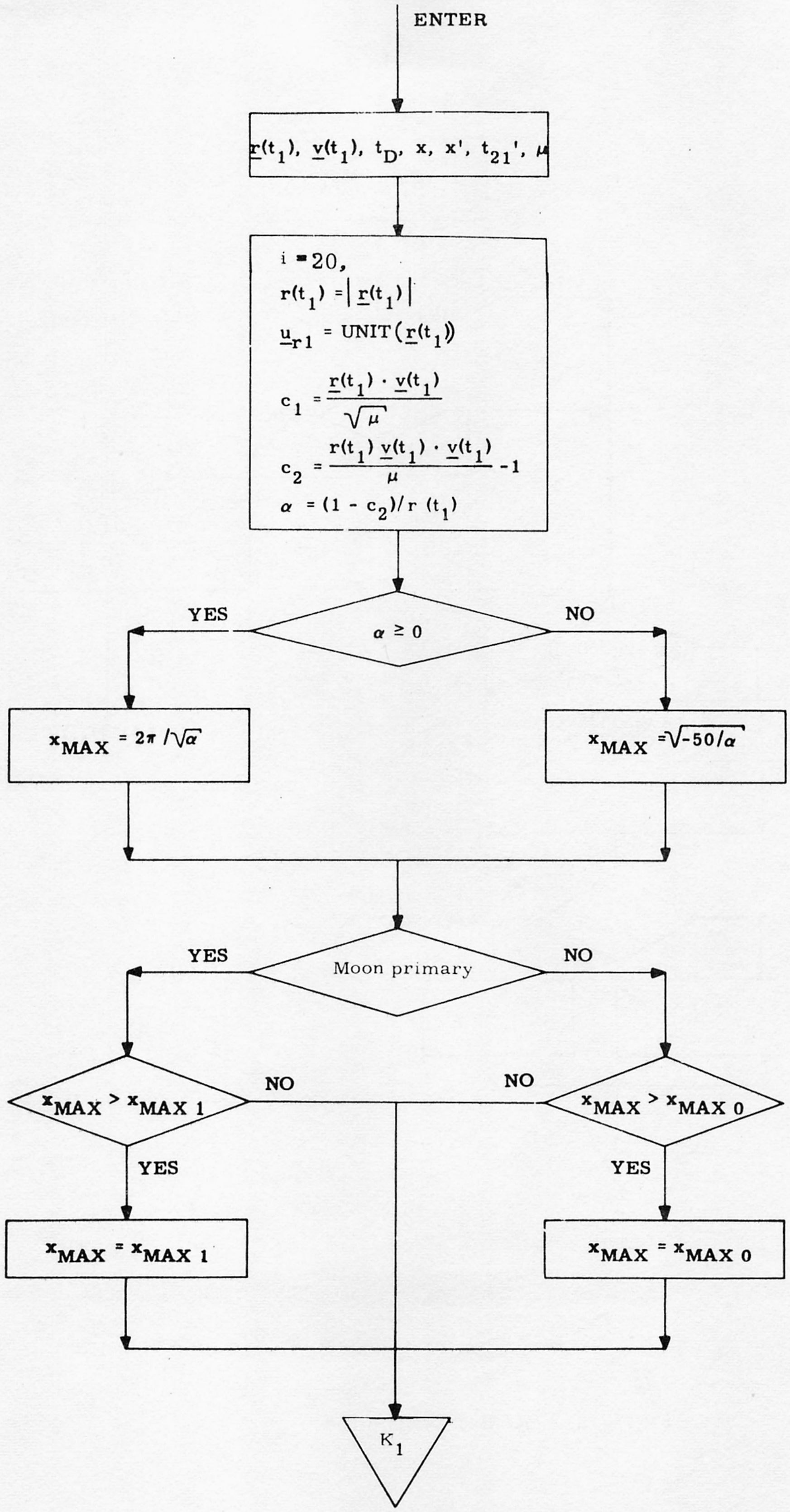


Figure 5. 5-1 Kepler Subroutine

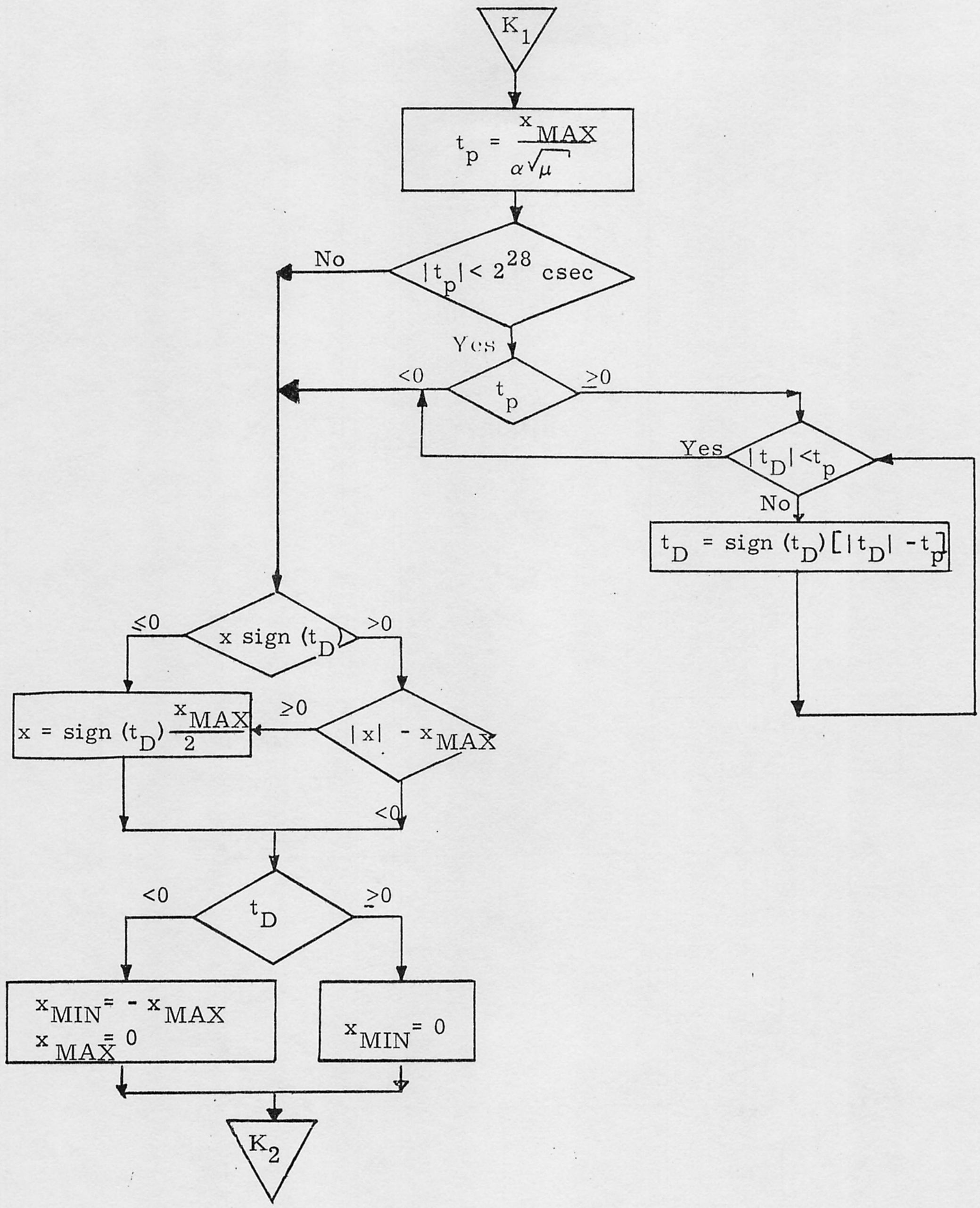


Fig. 5.5-2 Kepler Subroutine

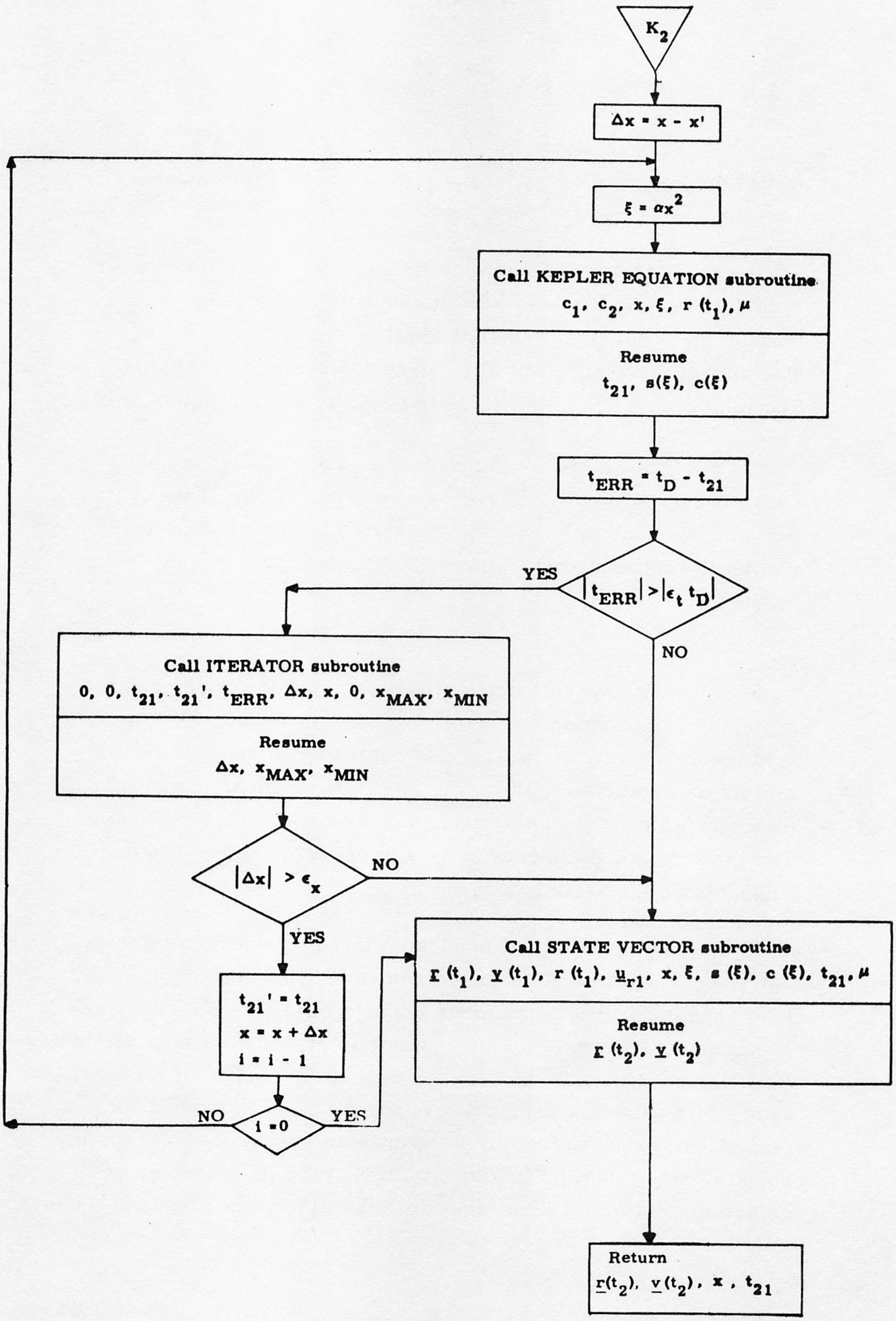


Fig. 5.5-3 Kepler Subroutine

5.5.6 LAMBERT SUBROUTINE

The Lambert Subroutine solves for the two body initial velocity vector given the initial and terminal position vectors and a transfer time between the two.

This section contains information to aid the reader in understanding the less obvious aspects of the Lambert Subroutine block diagrams depicted in Figs. 5.6-1 and 5.6-2. The subroutines referred to in these figures are presented in Section 5.5.10. Nomenclature is found in Section 5.5.1.2.

If the Lambert Subroutine is used repetitively and rapid computation is required, the previous value of the independent variable, cot γ, can be used as a starting point for the new iteration. Flag f_1 provides this option.

The Lambert Subroutine computes the normal to the trajectory, $\underline{u}_N$, using the two input position vectors. If these vectors are nearly colinear, it is desirable to specify the normal as an input rather than rely on the ill-defined normal based on the two input position vectors. Flag f_2 provides this option. The presence of the inputs in parentheses, therefore, is contingent upon the setting of these flags.

The theoretical bounds on the independent variable, cot γ, correspond to the infinite energy hyperbolic path and the parabolic path which closes through infinity. These bounds are dynamically reset by the iterator to provide a more efficient iteration scheme. In addition, if during the course of the iteration, cot γ causes a parameter of the problem to exceed its maximum as determined by its allowable range, the appropriate bound is reset and the iterator continues trying to find an acceptable solution. (This logic does not appear in Figs. 5.6-1 and 2

as it is pertinent only to fixed-point programming). If no acceptable solution is reached, the transfer time input was too small to produce a practical trajectory between the input position vectors. When this happens, $\Delta \cot \gamma$ approaches its granularity limit ϵ_c before time converges to within a fraction ϵ_t of the desired time. However, this same granularity condition exists when the sensitivity problem occurs as described in the Kepler Subroutine, Section 5.5.5. In this case an acceptable solution does exist. This dual situation is resolved via a third convergence criterion. If the error in transfer time is greater than the usual fraction ϵ_t of the desired transfer time, but still less than a slightly larger fraction k_1 of the desired transfer time and $\Delta \cot \gamma$ is less than ϵ_c, then the solution is deemed acceptable and the required velocity is computed. If, on the other hand, the error in the transfer time is greater than the fraction k_1 of the desired transfer time and $\Delta \cot \gamma$ is less than ϵ_c, then the flag f_5 is set to indicate that the required velocity will have degraded accuracy.

The maximum number of iterations allowable in the Lambert Subroutine has been made an input to the routine, in order to give the user direct control over the maximum computation time to be allowed. If no initial guess of $\cot \gamma_0$ is supplied, this number should be set to 20. If an initial guess of $\cot \gamma_n$ is supplied, which is sufficiently close to the true $\cot \gamma_0$, then this number may be set as low as 5.

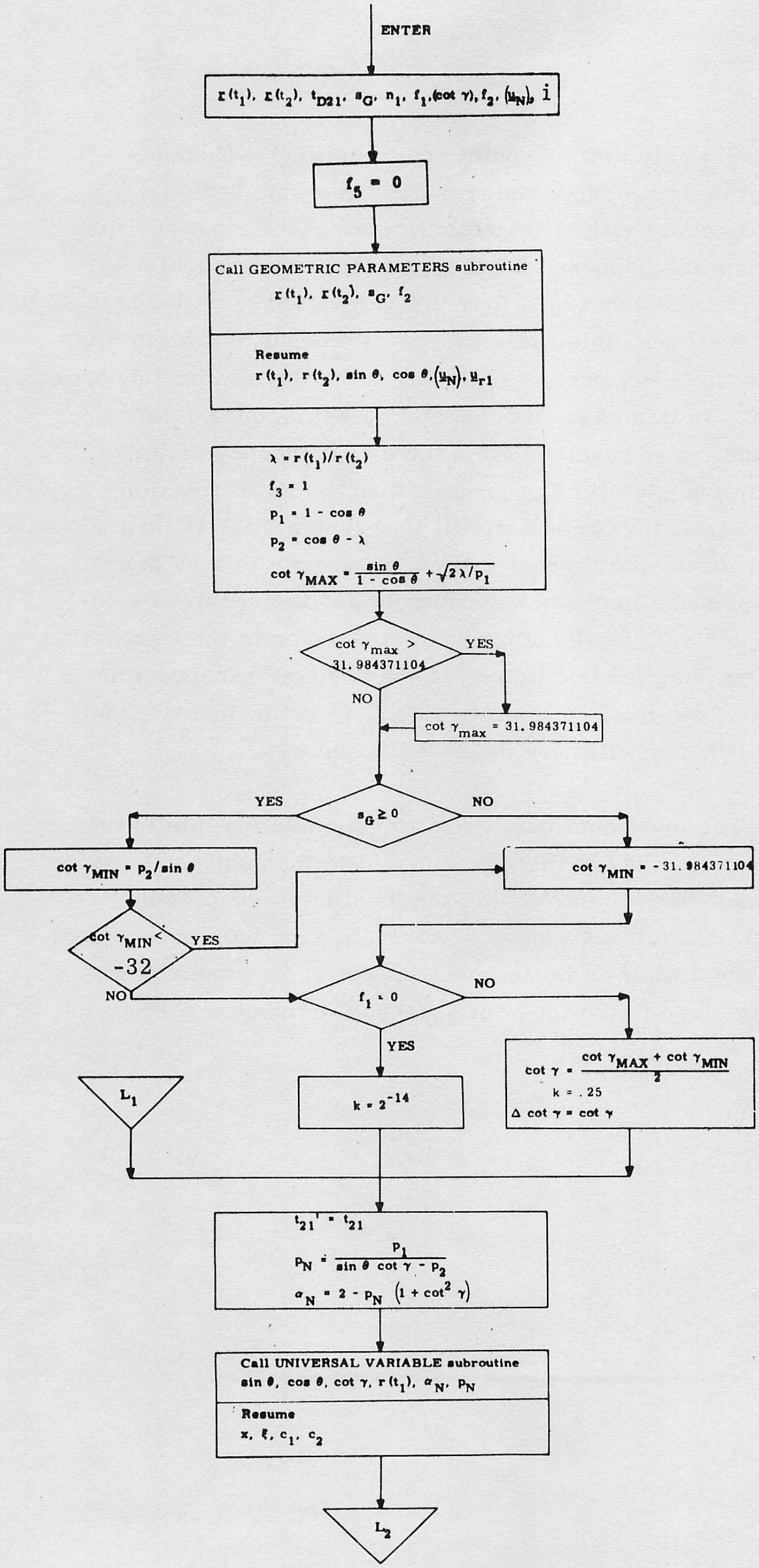


Figure 5.6-1 Lambert Subroutine

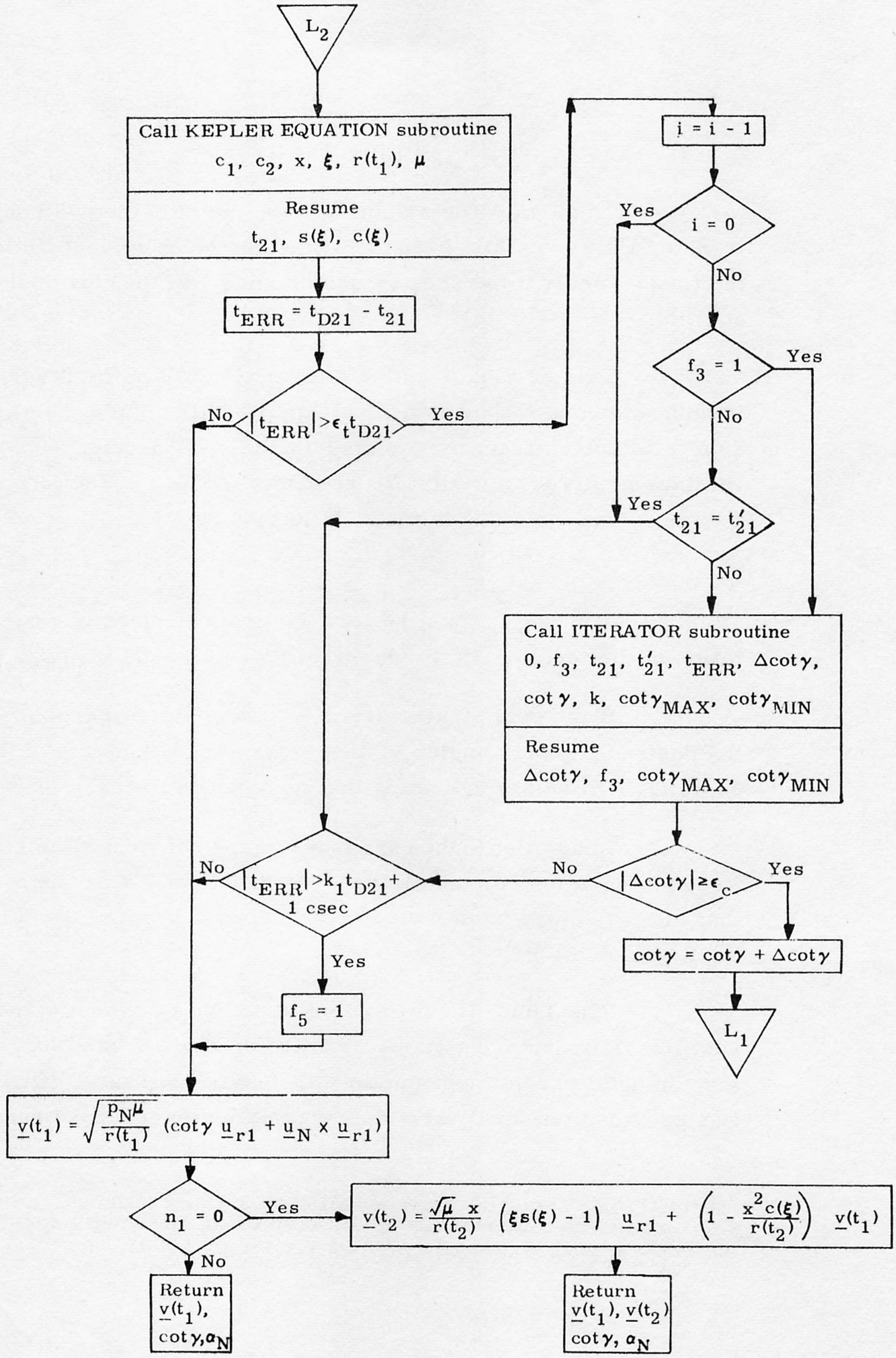


Figure 5.6-2 Lambert Subroutine

5.5.7 TIME-THETA SUBROUTINE

The Time-Theta Subroutine solves for the two body transfer time given the initial position and velocity vectors and the true anomaly difference (transfer angle) to the terminal position.

This section contains information to aid the reader in understanding the less obvious aspects of the Time-Theta Subroutine block diagram depicted in Fig. 5.7-1. The subroutines referred to in this figure are presented in Section 5.5.10. Nomenclature is found in Section 5.5.1.2.

The flag f_6 must be zero if the user desires computation of the terminal state vector in addition to the transfer time.

If the conic trajectory is a parabola or hyperbola and the desired transfer angle, θ, lies beyond the asymptote of the conic, f_7 will be set indicating that no solution is possible.*

In addition to the parameter range constraints imposed on Kepler's equation, the additional restriction on Time-Theta that the trajectory must not be near rectilinear is indicated by the range of cot γ.*

The Time-Theta problem is not well defined for near rectilinear trajectories, i e. the transfer angle θ is no longer a meaningful problem parameter. This will not cause difficulties provided the input variables are within the specified ranges.

---

*If the Time-Theta Routine is called with inputs for which no solution is possible (for either or both of these two reasons), the routine will abort with an alarm code of 20607.

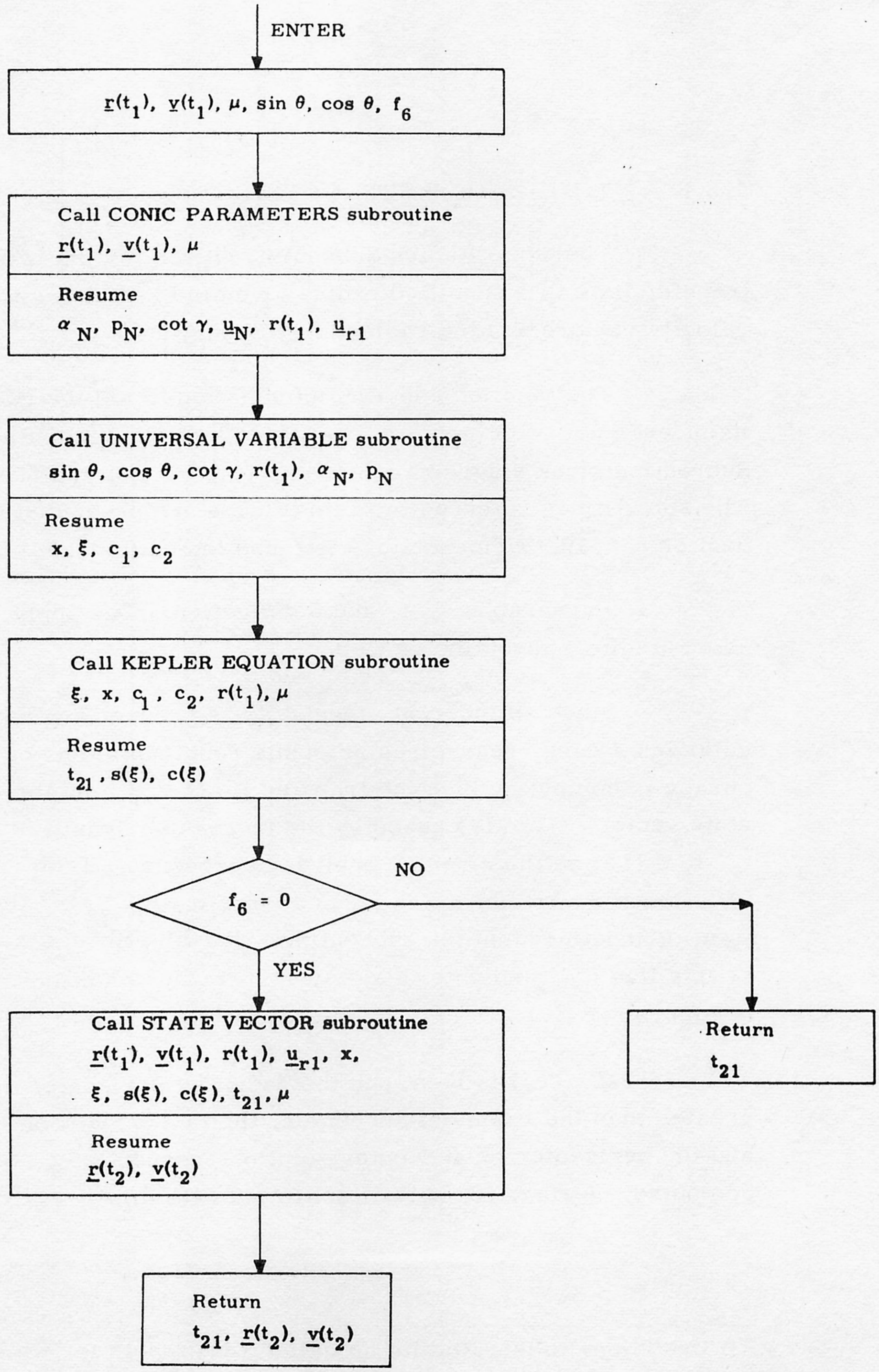


Figure 5.7-1 Time-Theta Subroutine

5.5.8 TIME-RADIUS SUBROUTINE

The Time Radius Subroutine solves for the two body transfer time to a specified radius given the initial position and velocity vectors and the radius magnitude.

This section contains information to aid the reader in understanding the less obvious aspects of the Time-Radius Subroutine block diagrams depicted in Figs. 5.8-1 and 5.8-2. The subroutines referred to in this figure are presented in Section 5.5.10. Nomenclature is found in Section 5.5.1.2.

Paragraphs 3, 4, and 5 of Section 5.5.7 apply to the Time-Radius Subroutine as well.*

Since an inherent singularity is present for the circular orbit case, near-circular orbits result in a loss of accuracy in computing both the transfer time, t_{21}, and the final state vector. This is caused by the increasing sensitivity of t_{21} to $r(t_2)$ as the circular orbit is approached. In the extreme case that eccentricity is less than approximately 2^{-18}, the problem is undefined and the subroutine will exit without a solution, setting flag f9 to indicate this.* (The precise conditions are given in Figure 5.8-1.)

If $r(t_2)$ is less than the radius of pericenter or greater than the radius of apocenter, then $r(t_2)$ will be ignored and the pericenter or apocenter solution, respectively, will be computed. A flag, f_8, will be set to indicate this.

---

*If the Time-Radius Routine is called with inputs for which no solution is possible (for any one or more of the reasons given in Paragraphs 4 and 5 of Section 5.5.7 and Paragraph 4 above), the routine will abort with an alarm code of 20607_8.

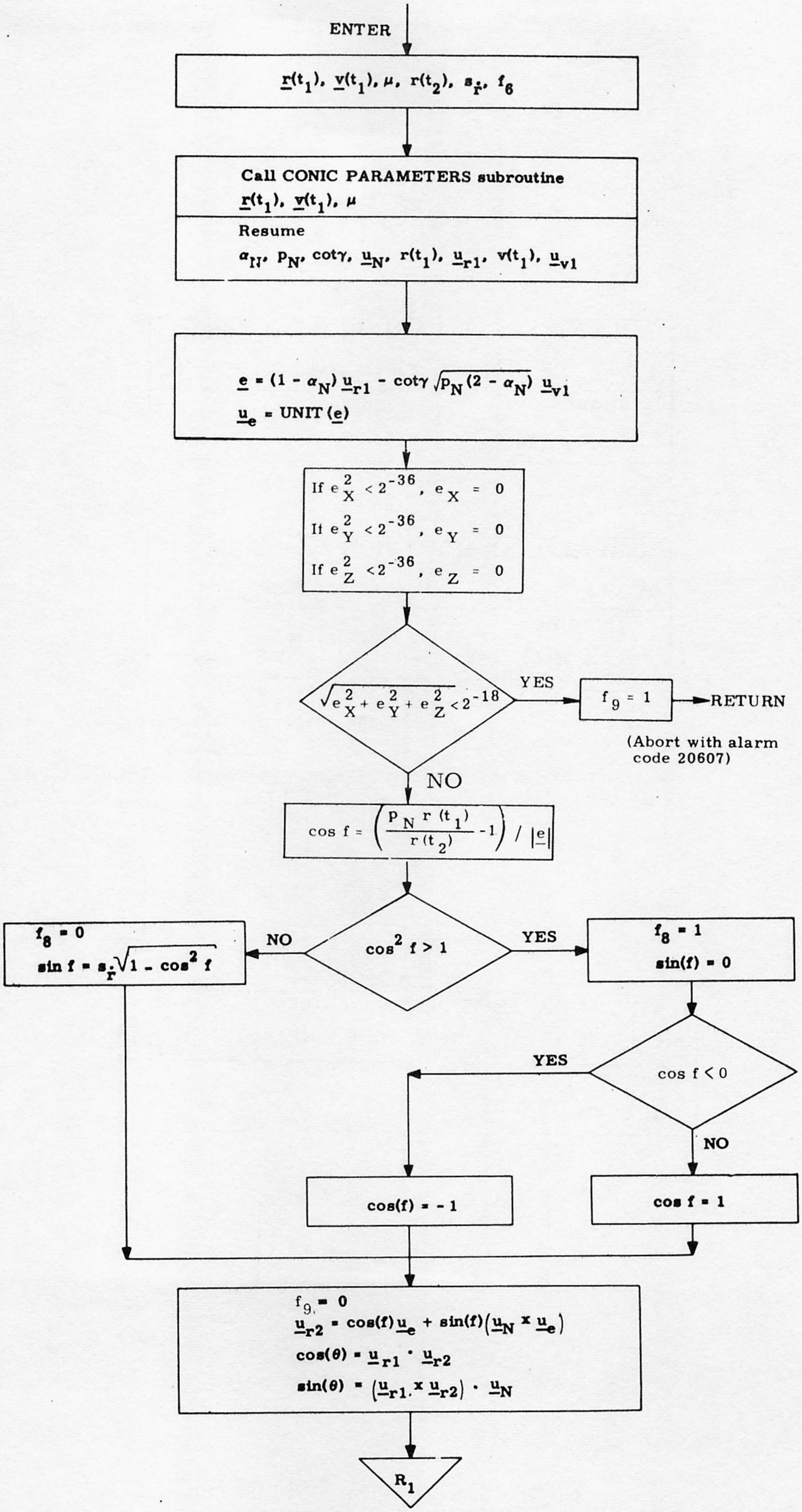


Figure 5.8-1. Time-Radius Subroutine

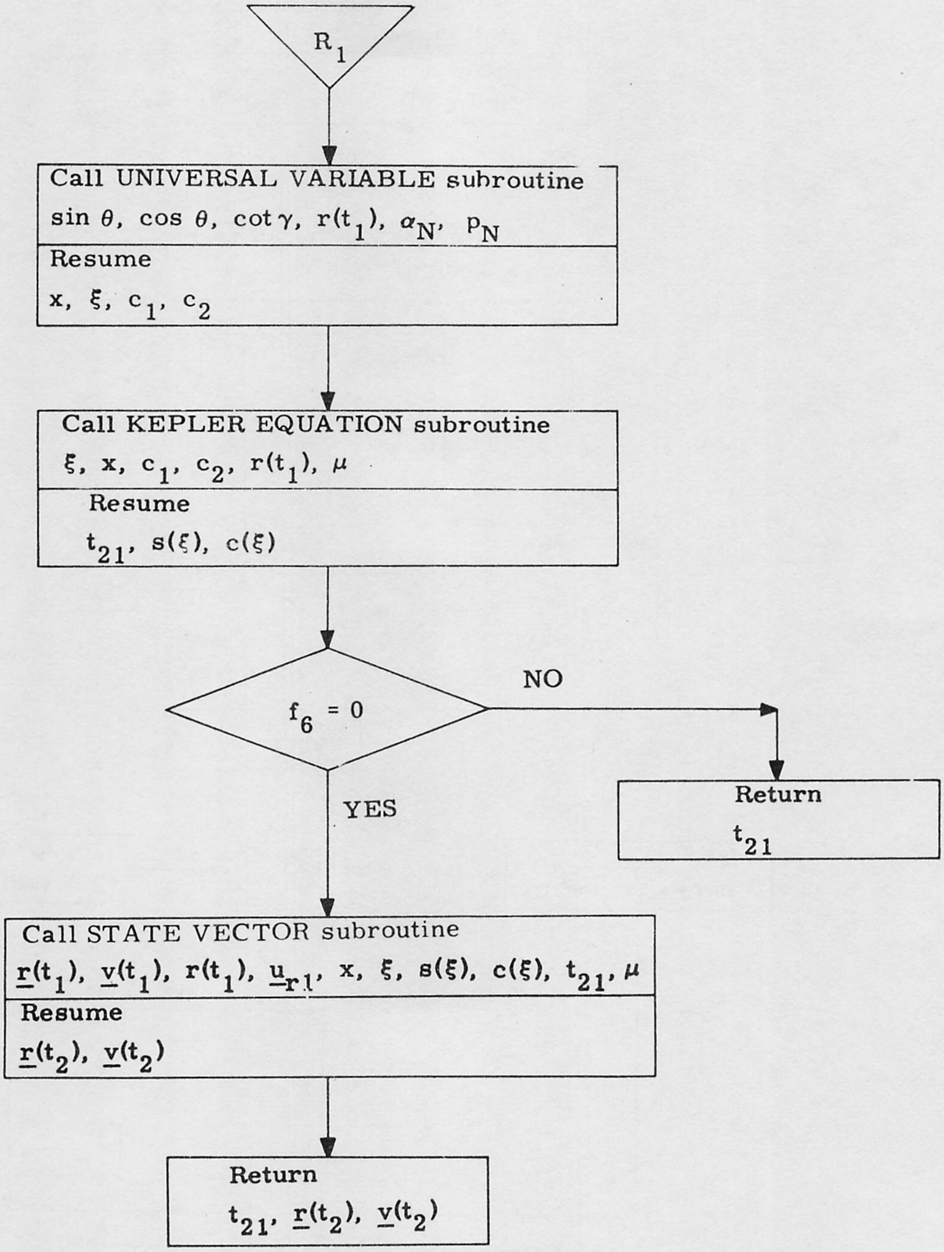


Figure 5.8-2 Time-Radius Subroutine

5.5.9 APSIDES SUBROUTINE

The Apsides Subroutine solves for the two body radii of apocenter and pericenter and the eccentricity of the trajectory given the position and velocity vectors for a point on the trajectory.

This subroutine is depicted in Fig. 5.9-1. The subroutines referred to in this figure are presented in Section 5.5.10 and the nomenclature is found in Section 5.5.1.2.

It is characteristic of this computation that the apsides become undefined as the conic approaches a circle. This is manifested by decreasing accuracy. When the conic is nearly parabolic, or hyperbolic, the radius of apocenter is not defined. In this event the radius of apocenter will be set to the maximum positive value allowed by the computer.

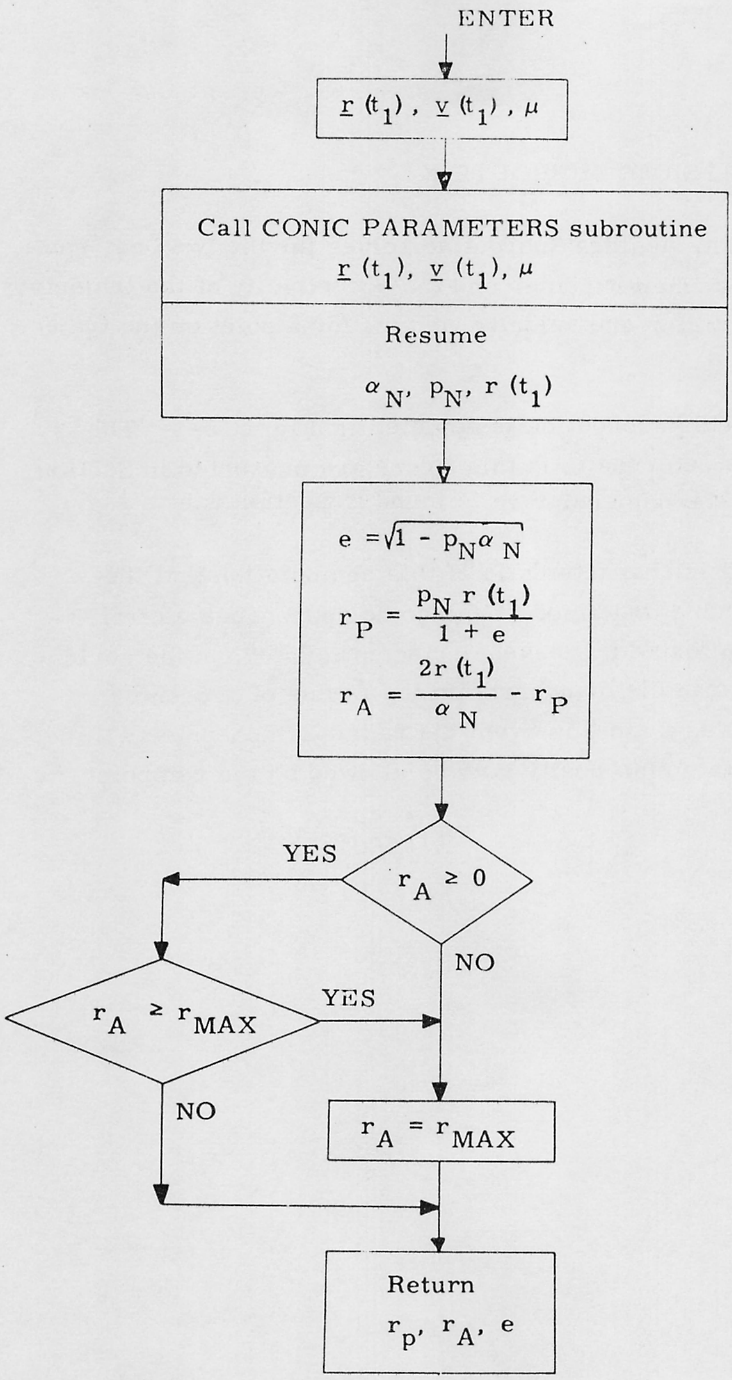


Figure 5.9-1 Apsides Subroutine

5.5.10 MISCELLANEOUS SUBROUTINES

There are, as part of the Conic Trajectory Subroutines, three subroutines which are useful in their own right. These are the Conic Parameters, the Geometric Parameters and the Iterator Subroutines which are depicted in Figs. 5.10-1, 5.10-2 and 5.10-3, respectively.

The Conic Parameters and Geometric Parameters Subroutines are self explanatory.

The Iterator Subroutine serves several purposes. It is used when flag f_4 is set to zero to solve for the value of the independent variable which drives the error in the dependent variable to zero, provided the function is monotonically increasing. To improve convergence for functions whose derivative changes rapidly, the limits are reset as shown in the block diagram.

With f_4 set to 1, the Iterator seeks a minimum of the function, provided the first derivative is single-valued between the limits. The inputs are redefined so that "y" is the derivative of the independent variable with respect to the dependent variable, and "x" is the value at which the derivative was computed or approximated. Since the desired value of y is zero, $y_{ERR} = -y$.

Since the Iterator uses the "Regula Falsi" technique, it requires two sets of variables to begin iteration. If only one set is available, flag f_3 must be set to 1, causing the iterator to generate the independent variable increment from a percentage of the full range.

In addition to the above subroutines there are three other subroutines of primary interest to the five basic conic subroutines described in Sections 5.5.5 to 5.5.9. These are the Universal Variable Subroutine, the Kepler Equation Subroutine, and the State Vector Subroutine shown in Figs. 5.10-4, 5.10-5 and 5.10-6, respectively.

The Universal Variable Subroutine is utilized by the Lambert, the Time-Theta and the Time-Radius Subroutines to compute the universal parameter x required for the time equation. There are two different formulations required according to the size of the parameter w.

If the input to the subroutine requires the physically impossible solution that the trajectory "close through infinity", the problem will be aborted, setting flag f_7.

The Kepler Equation Subroutine computes the transfer time given the variable x and the conic parameters.

The State Vector subroutine computes the position and velocity vectors at a point along the trajectory given an initial state vector, the variable x and the transfer time.

The final miscellaneous subroutine, the SETMU Subroutine, is depicted in Fig. 5.10-7. It sets μ to the appropriate primary body gravitational constant consistent with the estimated CSM or LM state vector as defined in Section 5.2.2.6.

ENTER

$\underline{r}(t_1)$, $\underline{v}(t_1)$, μ

Call GEOMETRIC PARAMETERS subroutine

$\underline{r}\ (t_1)$, $\underline{v}\ (t_1)$, +1, 0

Resume

$r\ (t_1)$, $v\ (t_1)$, $\sin\gamma$, $\cos\gamma$, $\underline{u}_N$, $\underline{u}_{r1}$, $\underline{u}_{v1}$

$$\cot\gamma = \cos\gamma / \sin\gamma$$

$$c_3 = \frac{r(t_1)\ v(t_1)^2}{\mu}$$

$$\alpha_N = 2 - c_3$$

$$p_N = c_3 \sin^2\gamma$$

Return

α_N, p_N, $\cot\gamma$, $\underline{u}_N$, $r\ (t_1)$,

$\underline{u}_{r1}$, $v\ (t_1)$, $\underline{u}_{v1}$

Figure 5.10-1 Conic Parameters Subroutine

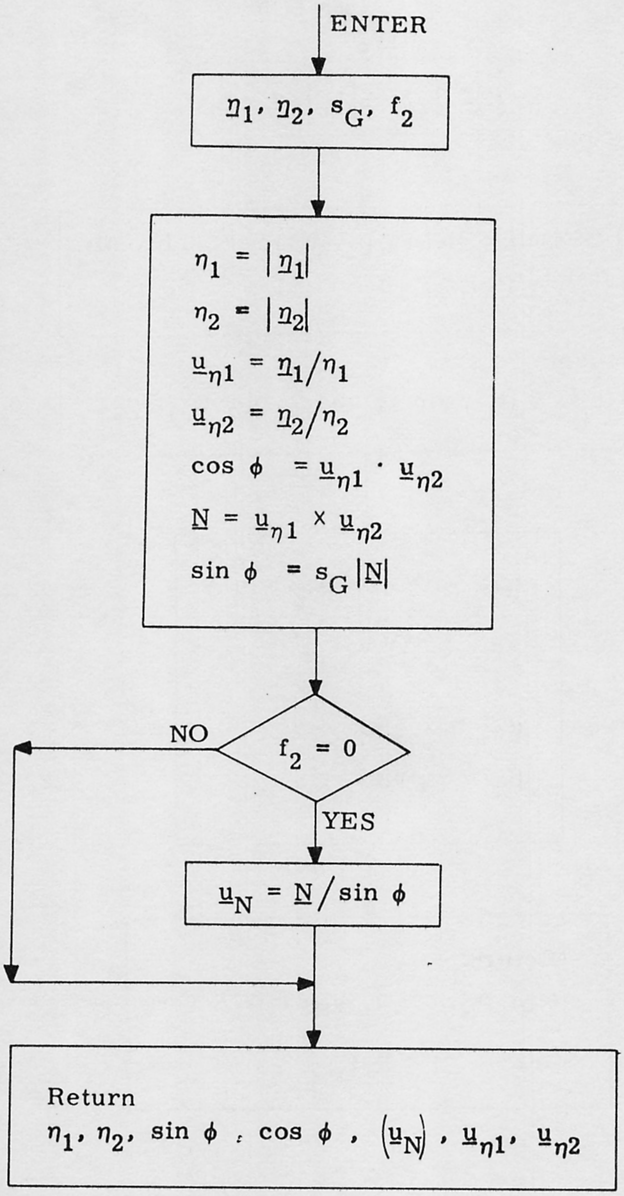


Figure 5.10-2 Geometric Parameters Subroutine

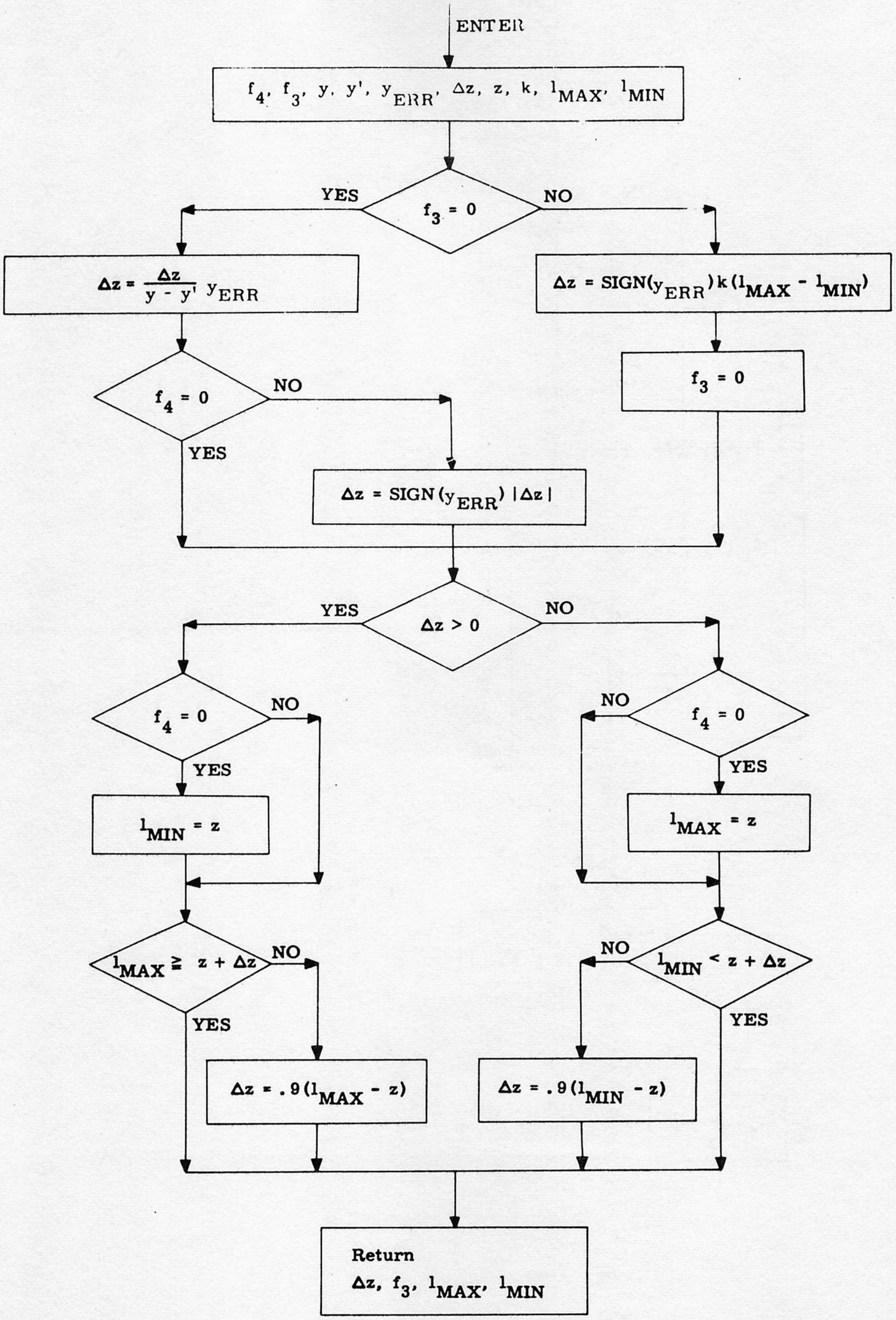


Figure 5.10-3 Iterator Subroutine

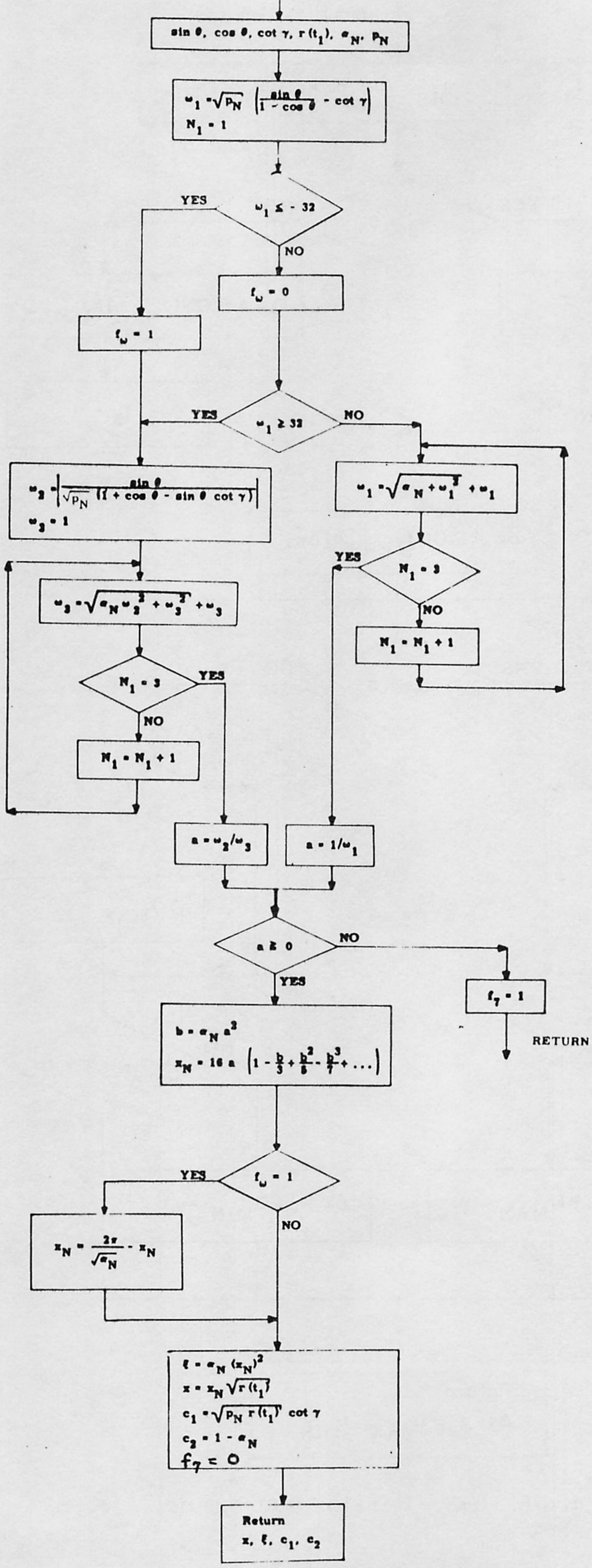


Figure 5.10-4 Universal Variable Subroutine

ENTER

$c_1,\ c_2,\ x,\ \xi,\ r(t_1),\ \mu$

$$s(\xi) = \frac{1}{3!} - \frac{\xi}{5!} + \frac{\xi^2}{7!} - \ldots$$

$$c(\xi) = \frac{1}{2!} - \frac{\xi}{4!} + \frac{\xi^2}{6!} - \ldots$$

$$t_{21} = \left[c_1 x^2 c(\xi) + x \left(c_2 x^2 s(\xi) + r(t_1) \right) \right] \Big/ \sqrt{\mu}$$

Return

$t_{21},\ s(\xi),\ c(\xi)$

Figure 5.10-5 Kepler Equation Subroutine

ENTER

$\underline{r}(t_1)$, $\underline{v}(t_1)$, $r(t_1)$, $\underline{u}_{r1}$, x, ξ, $s(\xi)$, $c(\xi)$, t_{21}, μ

$$\underline{r}(t_2) = \left(1 - \frac{x^2}{r(t_1)}\, c(\xi)\right) \underline{r}(t_1) + \left(t_{21} - \frac{x^3}{\sqrt{\mu}}\, s(\xi)\right) \underline{v}(t_1)$$

$$r(t_2) = \left|\underline{r}(t_2)\right|$$

$$\underline{v}(t_2) = \frac{\sqrt{\mu}\, x}{r(t_2)} \left(\xi\, s(\xi) - 1\right) \underline{u}_{r1} + \left(1 - \frac{x^2 c(\xi)}{r(t_2)}\right) \underline{v}(t_1)$$

Return

$\underline{r}(t_2)$, $\underline{v}(t_2)$

Figure 5.10-6 State Vector Subroutine

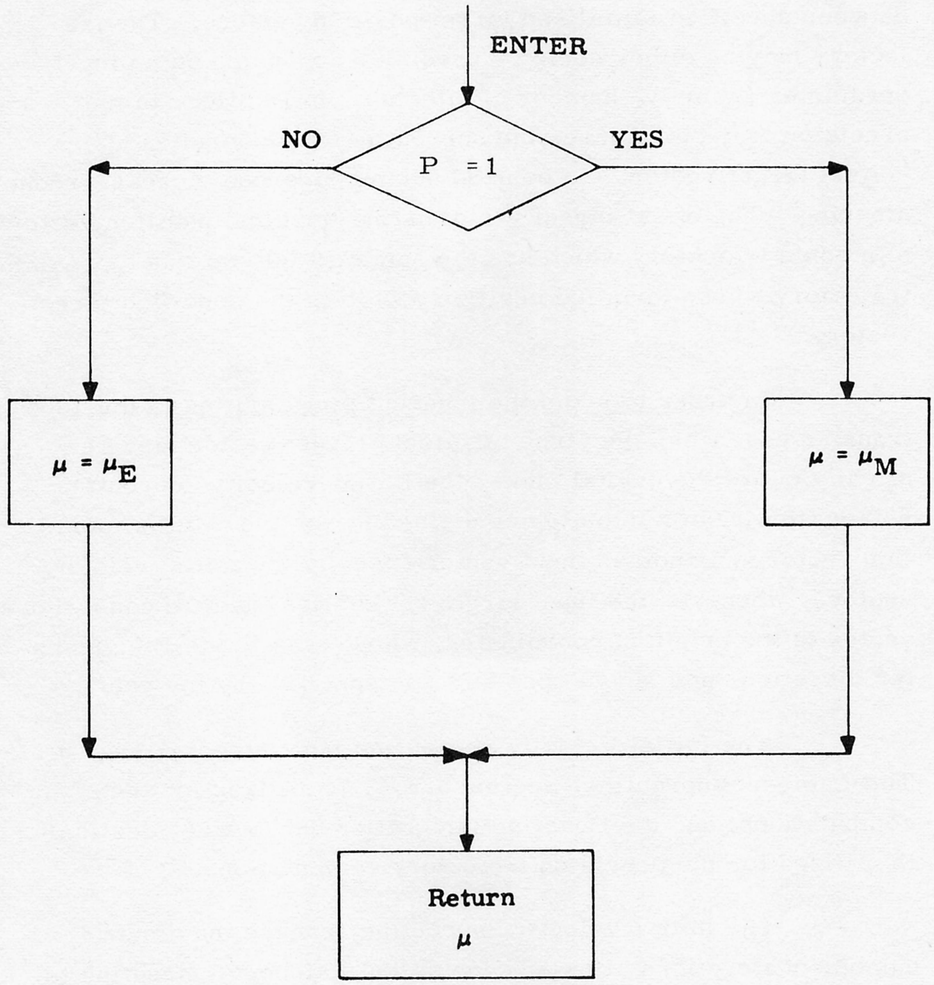


Figure 5. 10-7 SETMU Subroutine

5.5.11 INITIAL VELOCITY SUBROUTINE

The Initial Velocity Subroutine computes the required initial velocity vector for a trajectory of specified transfer time between specified initial and target position vectors. The trajectory may be either conic or precision depending on an input parameter (namely, number of offsets). In addition, in the precision trajectory case, the subroutine also computes an "offset target vector", to be used during pure-conic cross-product steering. The offset target vector is the terminal position vector of a conic trajectory which has the same initial state as a precision trajectory whose terminal position vector is the specified target vector.

In order to avoid the inherent singularities in the 180° transfer case when the (true or offset) target vector may be slightly out of the orbital plane, the Initial Velocity Subroutine rotates this vector into a plane defined by the input initial position vector and another input vector (usually the initial velocity vector), whenever the input target vector lies inside a cone whose vertex is the origin of coordinates, whose axis is the 180° transfer direction, and whose cone angle is specified by the user.

The Initial Velocity Subroutine is depicted in Fig. 5.11-1. The Lambert Subroutine, Section 5.5.6, is utilized for the conic computations; and the Coasting Integration Subroutine, Section 5.2.2, is utilized for the precision trajectory computations.

The Initial Velocity Subroutine sets the maximum number of iterations allowable inside the Lambert Subroutine to 20 in all cases.

Nomenclature for the Initial Velocity Subroutine

$\underline{r}(t_1)$	Initial position vector.
$\underline{v}(t_1)$	Vector (usually the actual initial velocity vector) used to determine whether the transfer from the initial position vector to the target vector is through a central angle of less or greater than 180^o, and also used in certain cases to specify the transfer plane (see text).
$\underline{r}_T(t_2)$	Target Vector (True target vector if $N_1 > 0$, or Offset target vector if $N_1 = 0$).
t_D	Desired transfer time from initial position vector to target vector.
N_1	Number of offsets to be used in calculating the offset target vector from the true target vector. ($N_1 = 0$ implies conic calculations only with offset target vector input).
ϵ	Cone Angle of a cone whose vertex is the coordinate origin and whose axis is the 180^o transfer direction (i.e., the negative initial position direction). The cone angle ϵ is measured from the axis to the side of the cone.
f_1	Switch set to 0 or 1 according to whether a guess of $\cot\gamma$ is input or not.
$[\cot\gamma]$	Guess of $\cot\gamma$.
$\underline{v}_T(t_1)$	Required initial velocity vector of a precision [a conic] trajectory which passes through the true [or offset] target vector, or the rotated true [or offset] target vector if the original target vector was in the cone, at the end of the desired transfer time, if $N_1 > 0$ [or $N_1 = 0$].
$\underline{r}(t_2)$	Computed offset target vector.
$\underline{v}_T'(t_2)$	Final precision [conic] velocity vector resulting from a precision [conic] update of the initial position vector and the required initial velocity vector $\underline{v}_T(t_1)$, if $N_1 > 0$, [or $N_1 = 0$, respectively].
$\underline{r}_T'(t_2)$	Final precision position vector.
$\cot\gamma$	Value to which the Lambert Subroutine converged (for later use as guess to minimize computation time).
f_2	Switch set to 0 or 1 according to whether the input (true or offset) target vector was not or was in the cone, and consequently was not or was rotated into the plane.
a	Semi-major axis.
s_G	$\begin{cases} 1 & \text{if transfer angle less than 180 deg.} \\ -1 & \text{if transfer angle greater than 180 deg.} \end{cases}$
$\underline{h}$	Unit normal to the trajectory.

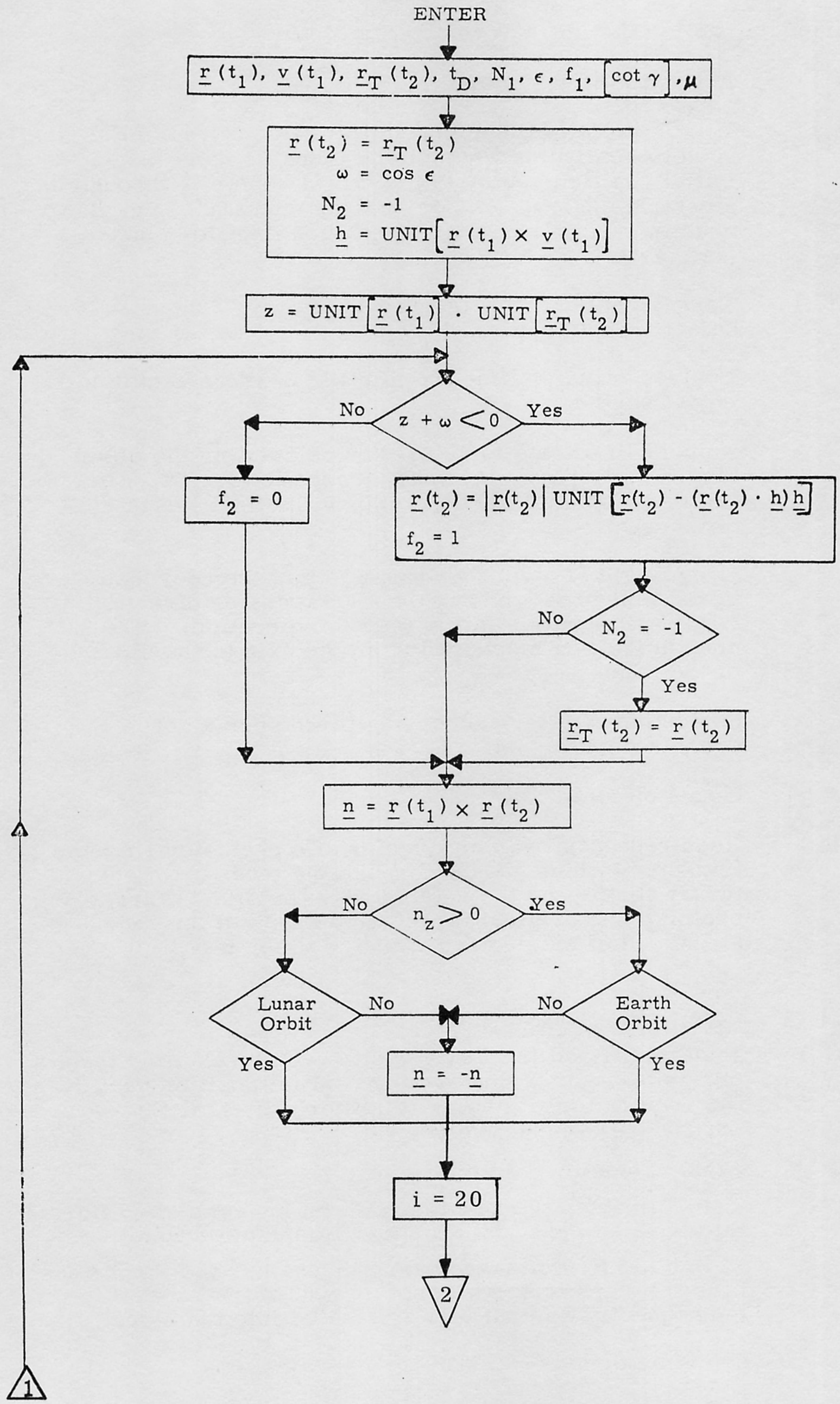


Figure 5.11-1 Initial Velocity Subroutine (page 1 of 2)

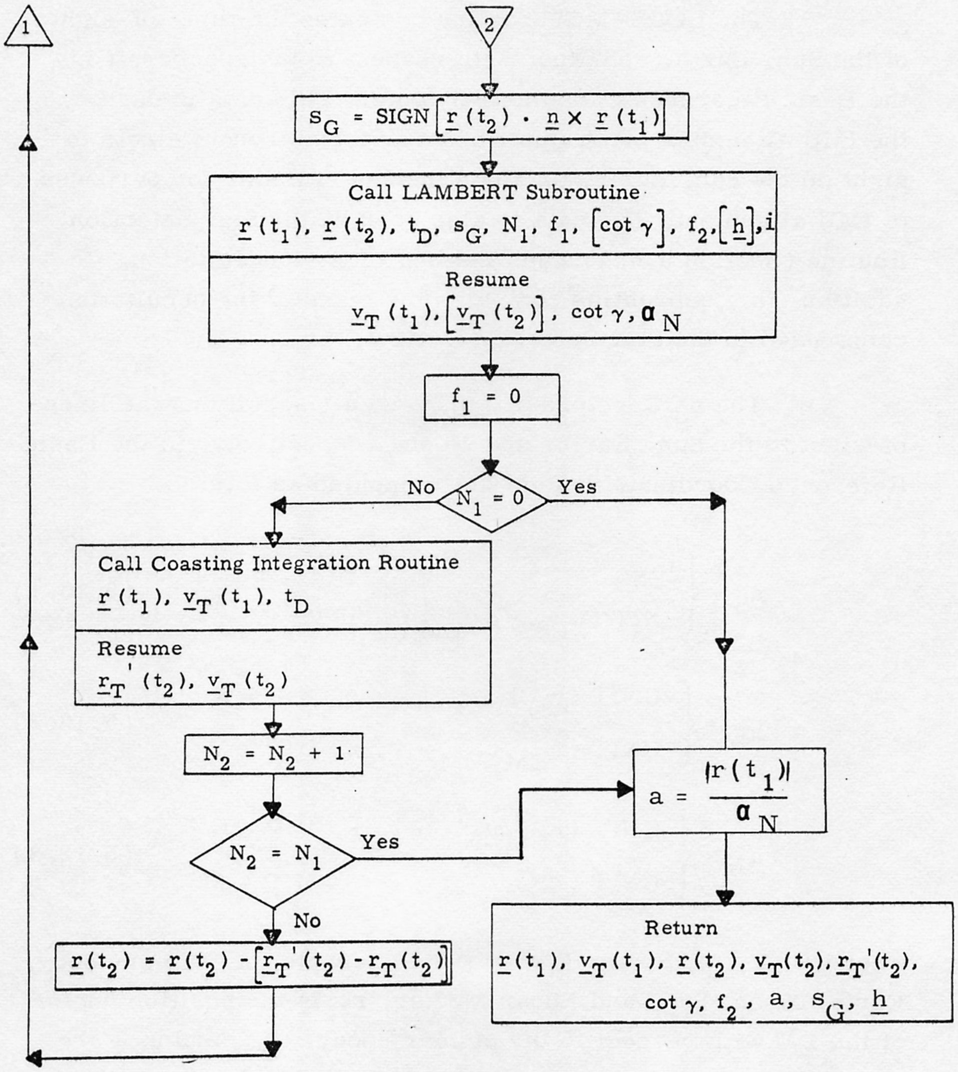


Figure 5.11-1 Initial Velocity Subroutine (page 2 of 2)

5.5.13 LOCSAM SUBROUTINE

The LOCSAM Subroutine computes the lines-of-sight of the Sun, Earth, and Moon with respect to the spacecraft in the Basic Reference Coordinate System. This data is used by the IMU alignment programs whenever the astronaut elects to sight on the Sun, Earth, or Moon instead of a star for purposes of IMU alignment. The data is also used by the Star Selection Routine (Section 5.6.4) when testing for star occultation. In addition, this subroutine computes the sizes of the occultation cones used in the Star Selection Routine.

The unit vectors $\underline{u}_S$, $\underline{u}_E$, and $\underline{u}_M$ specifying the lines-of-sight to the Sun, Earth, and Moon, respectively, in the Basic Reference Coordinate System are computed as follows:

$$\underline{u}_S = \begin{cases} \underline{u}_{ES} & \text{if } P = E \\ \text{UNIT}\,(\underline{u}_{ES} - \rho\, \underline{u}_{EM}) & \text{if } P = M \end{cases} \tag{5.13.1}$$

$$\underline{u}_E = \begin{cases} -\text{UNIT}\,(\underline{r}_L) & \text{if } P = E \\ -\text{UNIT}\,(r_{EM}\underline{u}_{EM} + \underline{r}_L) & \text{if } P = M \end{cases} \tag{5.13.2}$$

$$\underline{u}_M = \begin{cases} \text{UNIT}\,(r_{EM}\underline{u}_{EM} - \underline{r}_L) & \text{if } P = E \\ -\text{UNIT}\,(\underline{r}_L) & \text{if } P = M \end{cases} \tag{5.13.3}$$

where P, E, M, S, and L respectively denote the primary body, Earth, Moon, Sun, and Lunar Module, $\underline{r}_L$ is the position vector of the LM with respect to the primary body, $\underline{u}_{EM}$ and $\underline{u}_{ES}$ are the unit position vectors of the Moon and Sun with respect to the Earth obtained from the Lunar and Solar Ephemerides Subroutine of Section 5.5.4, r_{EM} is the mean distance (384402 km) between the Earth and Moon, and ρ is the ratio (0.00257125) of r_{EM} over the mean distance between the Sun and Earth. The line-of-sight vectors are determined for a time specified by the calling program or routine. When LOCSAM is used by the sighting mark routines it is not called until just after the astronaut finishes his optical marks on the Sun, Earth, or Moon, and the time used in this case is that recorded for the last mark. When LOCSAM is used by the Auto Optics Positioning

Routine (R-52) it is called just before positioning the spacecraft and the specified time is the present time. When LOCSAM is called by the IMU Realignment Program (P-52) just prior to calling the Star Selection Routine, the specified time is the present time plus an additional amount (T_{SS}) in order to project it into the middle of the sighting mark process. This is done to insure that the line-of-sight vector of the primary body used in the occultation test of the Star Selection Routine is that which occurs during the sighting mark process.

The occultation cones used in the Star Selection Routine for the Sun, Earth, and Moon are computed as follows:

$$c_S = \cos 60^o \tag{5.13.4}$$

$$c_E = \begin{cases} \cos\left[5^o + \sin^{-1}\left(\dfrac{R_E}{r_L}\right)\right] & \text{if } P = E \\ \cos 5^o & \text{if } P = M \end{cases} \tag{5.13.5}$$

$$c_M = \begin{cases} \cos 5^o & \text{if } P = E \\ \cos\left[5^o + \sin^{-1}\left(\dfrac{R_M}{r_L}\right)\right] & \text{if } P = M \end{cases} \tag{5.13.6}$$

where c is the cosine of one half the total angular dimension of a cone and represents a more convenient way of treating the dimension of a cone in the Star Selection Routine, r_L is the magnitude of the LM position vector, R_E is the equatorial radius (6378.166 km) of the Earth, and R_M is the mean radius (1738.09 km) of the Moon.

5.5.14 <u>PERICENTER - APOCENTER (PERIAPO) SUBROUTINE</u>

The Pericenter - Apocenter Subroutine computes the two body apocenter and pericenter altitudes given the position and velocity vectors for a point on the trajectory and the primary body.

This subroutine is depicted in Fig. 5.14-1.

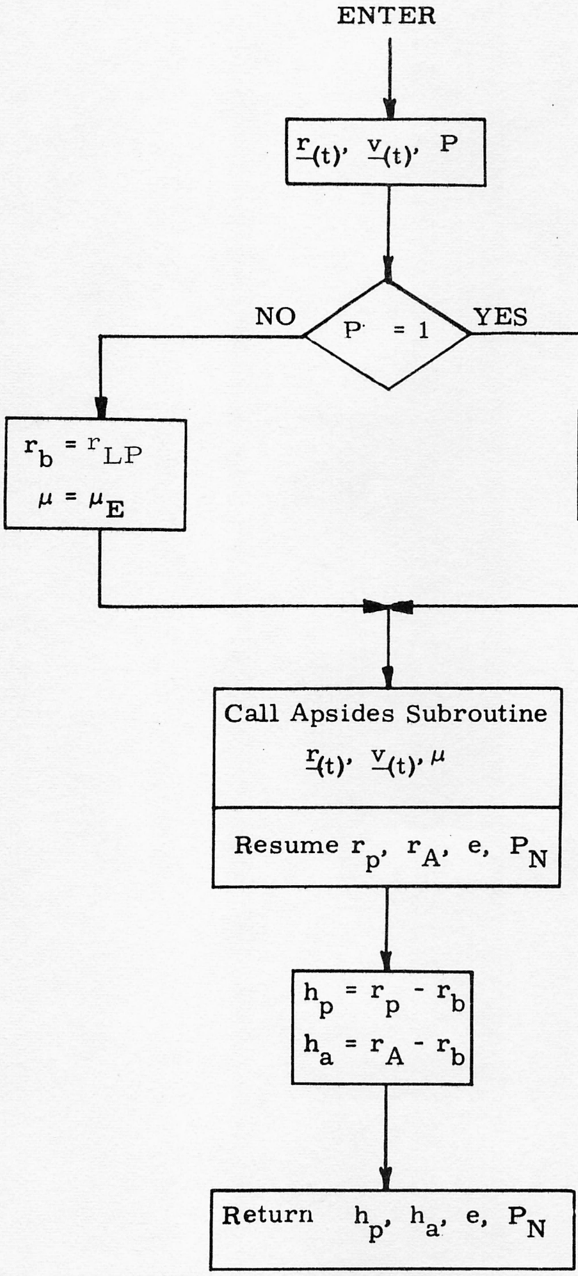


Figure 5.14-1 PERICENTER - APOCENTER SUBROUTINE

5.6 GENERAL SERVICE ROUTINES

5.6.1 GENERAL COMMENTS

The routines presented in this section are used for the following general service functions:

1) IMU alignment modes

2) Basic Coordinate Transformations

3) Computer initialization procedures

4) Special display routines which can be called by the astronaut

5.6.2 IMU ALIGNMENT MODES

5.6.2.1 Orbital Alignment

5.6.2.1.1 IMU Orientation Determination Program

The IMU Orientation Determination Program (P-51) is used during free-fall to determine the present IMU stable member orientation with respect to the Basic Reference Coordinate System by sighting on two navigation stars or known celestial bodies with the Alignment Optical Telescope (AOT) or the Crew Optical Alignment Sight (COAS). At the start of program P-51 the astronaut acquires the desired celestial bodies by maneuvering the spacecraft until they are visible in the optical device which he intends to use. During the above operation he checks to insure that gimbal lock does not occur. Afterwards, the AOT Mark Routine (R-53) is used twice to sight upon each of two celestial bodies. Prior to sighting on each body, the astronaut indicates to the computer which optical device he is going to use and which celestial body he is going to sight on. The method of selection and operation of the AOT and COAS is given in Section 5. 6. 3. 1. 1 and in the AOT Mark Routine (R-53) of Section 4. The identity of the celestial body is indicated by the astronaut at the beginning of the AOT Mark Routine by use of a star (or celestial body) code. Separate codes are provided for the 37 navigation stars, Sun, Earth, Moon, and a general celestial body. The latter code is referred to as the Planet code. If the astronaut selects the Sun, Earth, or Moon code, the subroutine LOCSAM of Section 5. 5. 13 will be called after optical sightings have been made on the body to compute the line-of-sight vector to that body in the Basic Reference Coordinate System. If he selects the Planet code, the computer will request after the optical sightings that he load in the coordinates of the desired celestial body in the Basic Reference Coordinate System.

After optical sightings have been made on both celestial bodies, the computer has the unit line-of-sight vectors for the two bodies in both IMU stable member and basic reference coordinates. Let $\underline{u}'_{CBA}$ and $\underline{u}'_{CBB}$ be the unit vectors in IMU stable member coordinates for the two celestial bodies (A and B) which are computed as shown in Section 5.6.3.1.1, and let $\underline{u}_{CBA}$ and $\underline{u}_{CBB}$ be the unit vectors for the same bodies in basic reference coordinates which are either taken from the computer's star catalog in fixed memory, computed by the subroutine LOCSAM, or loaded by the astronaut. At this point in the program the Sighting Data Display Routine (R-54 of Section 4) computes the angle ϕ' between the unit line-of-sight vectors ($\underline{u}'_{CBA}$ and $\underline{u}'_{CBB}$) obtained for the two bodies in stable member coordinates and the angle ϕ between the corresponding unit line-of-sight vectors ($\underline{u}_{CBA}$ and $\underline{u}_{CBB}$) for the two bodies in basic reference coordinates. The angular difference $\phi - \phi'$ is displayed to the astronaut and he either accepts the results or repeats the IMU orientation determination process.

If he accepts the results of the Sighting Data Display Routine, the unit vectors $\underline{u}'_{CBA}$, $\underline{u}'_{CBB}$, $\underline{u}_{CBA}$, and $\underline{u}_{CBB}$ are used to determine the present stable member orientation and REFSMMAT, using the procedure given in Section 5.6.3.4.1.

5.6.2.1.2 IMU Realign Program

The IMU Realign Program (P-52) is used during free-fall to re-align the IMU to its presently assumed orientation or to align it from a known orientation to one of the desired orientations given in Section 5.6.3.4 and in P-52 of Section 4. This alignment is made by sighting on two navigation stars or known celestial bodies with the AOT or the COAS.

At the beginning of program P-52 the astronaut indicates which of the following stable member orientations is desired:

1) Preferred - for thrusting maneuvers
2) Landing Site - for LM lunar landing or launch
3) Nominal - for alignment with respect to local vertical
4) REFSMMAT - for re-alignment to presently assumed orientation

The Preferred, Landing Site, and Nominal orientations are defined in Section 5.6.3.4. If the astronaut selects the Landing Site or Nominal orientation, it is computed by program P-52 in the manner shown in Section 5.6.3.4 and in P-52 of Section 4. The Preferred orientation must be computed prior to entering program P-52. Whenever the astronaut selects the Preferred, Landing Site, or Nominal orientation, the program also computes and displays the IMU gimbal angles for the desired stable member orientation using the present vehicle attitude. These angles are computed by the routine CALCGA of Section 5.6.3.2.2 where the inputs to this routine are the unit vectors ($\underline{u}_{XSM}$, $\underline{u}_{YSM}$, $\underline{u}_{ZSM}$) defining the desired stable member axes with respect to the Basic Reference Coordinate System, and the unit vectors ($\underline{x}_{NB}$, $\underline{y}_{NB}$, $\underline{z}_{NB}$) defining the present navigation base axes with respect to the Basic Reference Coordinate System, which are computed as follows:

$$\underline{x}_{NB} = [\text{ REFSMMAT}]^T \, \underline{x}'_{NB}$$

$$\underline{y}_{NB} = [\text{ REFSMMAT}]^T \, \underline{y}'_{NB}$$

$$\underline{z}_{NB} = [\text{ REFSMMAT}]^T \, \underline{z}'_{NB}$$

where [REFSMMAT]^T is the transpose of the present [REFSMMAT] and $\underline{x}'_{NB}$, $\underline{y}'_{NB}$, and $\underline{z}'_{NB}$ define the navigation base axes with respect to the present Stable Member Coordinate System and are computed by the routine CALCSMSC of Section 5.6.3.2.5.

If the computed IMU gimbal angles are unsatisfactory, the astronaut maneuvers the vehicle to a more suitable attitude and has the program re-compute and display the new gimbal angles. Once satisfactory angles have been obtained, the astronaut keys in a "PROCEED" and is then requested to indicate whether the IMU is to be aligned to the desired orientation by use of the Coarse Align Routine (R-50 of Section 4) or by torquing the gyros. If he elects to have the IMU gyro torqued to the desired orientation, the gyro torquing angles are computed by the routine CALCGTA of Section 5.6.3.2.3 where the inputs to this routine are the unit vectors ($\underline{x}_D$, $\underline{y}_D$, $\underline{z}_D$) defining the desired stable member axes with respect to the present Stable Member Coordinate System which are computed as follows:

$$\underline{x}_D = [\text{REFSMMAT}] \ \underline{u}_{XSM}$$

$$\underline{y}_D = [\text{REFSMMAT}] \ \underline{u}_{YSM}$$

$$\underline{z}_D = [\text{REFSMMAT}] \ \underline{u}_{ZSM}$$

where $\underline{u}_{XSM}$, $\underline{u}_{YSM}$, and $\underline{u}_{ZSM}$ were previously defined. During the period when the gyros are being torqued the IMU gimbal angles are displayed to the astronaut so that he may avoid gimbal lock by maneuvering the spacecraft. After the gyro torquing process the astronaut either terminates program P-52 or performs a fine alignment with optical sightings on two celestial bodies.

If the astronaut elects to have the IMU aligned to the desired orientation by means of the Coarse Align Routine (R-50) this routine will command the IMU to the computed gimbal angles if any of the required gimbal angle changes is greater than one degree. If all of the required gimbal angle changes are less than or equal to one degree, routine R-50 will leave the IMU at its present orientation and it will be assumed by program P-52 that it is at the desired orientation. It should be noted that there is no computation of gimbal angles or coarse alignment of the IMU if the astronaut selects the REFSMMAT orientation at the beginning of program P-52.

At this point in program P-52 the Inflight Fine Align Routine (R-51 of Section 4) is used to perform the necessary optical sightings for fine alignment. This routine performs the alignment by using various other routines. Initially, the astronaut decides whether to use R53, the AOT Mark Routine of Section 5. 6. 3 (nominal procedure), or to use the Lunar Surface Sighting Mark Routine (R59). (Procedures for R59 are described in Section 5. 6. 3. 1. 2.) If he selects the normal inflight procedures, Routine R-51 will call the subroutine LOCSAM of Section 5. 5. 13 prior to calling the Star Selection Routine since LOCSAM computes certain parameters which are used in the occultation tests of the Star Selection Routine. If the Star Selection Routine is unable to find two satisfactory stars at the present vehicle attitude, the astronaut either repeats the above process of changing the vehicle attitude and using the Star Selection Routine or selects his own celestial bodies. It should be noted that the Star Selection Routine only selects stars for the forward viewing position of the AOT. However, the astronaut may select stars or celestial bodies with the intention of using one of the other viewing positions described in Section 5. 6. 3. 1. 1.

Prior to performing the optical sightings on each celestial body, the astronaut is given the option of either maneuvering the vehicle himself to place the celestial body at the desired sighting position or having the Automatic Optics (LM) Positioning Routine (R-52 of Section 4) command the vehicle attitude so as to place the celestial body at the desired sighting position. The sighting position is established by the astronaut at the beginning of the Automatic Optics (LM) Positioning Routine in the same manner as is done at the start of the Inflight Sighting Mark Routine. By selecting the appropriate detent code as described in Section 5.6.3.1.1 and in routines R-52 and R-53 of Section 4, the sighting position can be the center of the field-of-view of the COAS or any of the six fields-of-view of the AOT. In addition to establishing the sighting position, the astronaut can also specify the celestial body (star, etc.) which is to be used in the Automatic Optics (LM) Positioning Routine just as is described for the AOT Mark Routine in Section 5.6.2.1.1. If he selects the Sun, Earth or Moon Code, the subroutine LOCSAM is called prior to positioning the spacecraft to compute the line-of-sight vector to that body. If he selects the Planet code, the computer will request that he load in the coordinates of the desired celestial body just before positioning the spacecraft. Once the celestial body is properly positioned in the field-of-view, the AOT Mark Routine is used to perform the sightings on the body.

After sightings have been made on two celestial bodies, the computer has the unit line-of-sight vectors for the two bodies in both IMU stable member and basic reference coordinates. Let $\underline{u}_{CBA}$ and $\underline{u}_{CBB}$ be the two unit vectors in IMU stable member coordinates for the two celestial bodies (A and B) which are computed as shown in Section 5. 6. 3. 1. 1, and let $\underline{u}''_{CBA}$ and $\underline{u}''_{CBB}$ be the unit vectors for the same bodies in basic reference coordinates which are either taken from the computer's star catalog in fixed memory, computed by the subroutine LOCSAM, or loaded by the astronaut. The vectors $\underline{u}''_{CBA}$ and $\underline{u}''_{CBB}$ are next transformed to the desired IMU Stable Member Coordinate System as follows:

$$\underline{u}'_{CBA} = [REFSMMAT]_D \ \underline{u}''_{CBA}$$

$$\underline{u}'_{CBB} = [REFSMMAT]_D \ \underline{u}''_{CBB}$$

where $[REFSMMAT]_D$ is the matrix (Section 5. 6. 3. 4) for transforming vectors from the Basic Reference Coordinate System to the desired IMU Stable Member Coordinate System. At this point in the program the Sighting Data Display Routine (R-54) of Section 4) computes the angle ϕ between $\underline{u}_{CBA}$ and $\underline{u}_{CBB}$, and the angle ϕ' between $\underline{u}'_{CBA}$ and $\underline{u}'_{CBB}$ and displays the angular difference $\phi - \phi'$ to the astronaut. If he accepts the results, the four vectors are used in the routine AXISGEN of Section 5. 6. 3. 2. 4 to compute the desired

stable member axes with respect to the present Stable Member Coordinate System, which are used in the Gyro Torquing Routine (R-55 of Section 4). Routine R-55 computes the gyro torquing angles required to drive the IMU stable member to the desired orientation by using the above vectors in the routine CALCGTA of Section 5.6.3.2.3. The gyro torquing angles are displayed to the astronaut so that he can decide whether to have the gyros torqued through these angles or not. If he is not satisfied with the results of the Sighting Data Display Routine or the Gyro Torquing Routine, he may repeat the optical sightings without having to terminate the program.

It should be noted that the ground can indicate to the LGC via uplink the present stable member orientation or a desired stable member orientation. If the present orientation is being indicated, this is done by transmitting a REFSMMAT to the LGC. Under normal circumstances, however, it would not be desirable for the ground to indicate the present stable member orientation since this orientation should be determined by the PGNCS. However, if an orientation different from the present orientation is desired by the ground, this desired orientation can be transmitted as a Preferred orientation. By treating a desired stable member orientation in this manner, program P-52 will be able to correct for any large differences between the present and desired orientations by coarse alignment. In addition, this approach avoids the introduction of orientation errors which affect celestial body acquisition.

5.6.2.2 Lunar Surface Alignment

5.6.2.2.1 General

There are several methods or techniques of IMU alignment available to the astronaut in the Lunar Surface Align Program (P-57). These techniques are denoted as Alignment Techniques 0 through 3.

In order to align the IMU stable member to a desired orientation with respect to the Basic Reference Coordinate System with any of the four techniques, it is necessary to determine two separate line-of-sight (LOS) directions with respect to both the Basic Reference Coordinate System and the present IMU Stable Member Coordinate System. It should be noted that the present orientation of the IMU Stable Member Coordinate System with respect to the Basic Reference Coordinate System does not have to be known in order to perform the alignment. The two directions in both of the above coordinate systems are specified by four unit LOS vectors where $\underline{s}_A$ and $\underline{s}_B$ denote the unit vectors for the directions A and B in present stable member coordinates, and $\underline{s}_A''$ and $\underline{s}_B''$ are the unit vectors for the corresponding directions in basic reference coordinates. For example, if A and B are two navigation stars and the alignment is made with technique 2 of P-57, the unit vectors $\underline{s}_A$ and $\underline{s}_B$ in present stable-member coordinates would be those obtained by optical sightings with the AOT, and $\underline{s}_A''$ and $\underline{s}_B''$ would be the unit vectors for A and B in basic reference coordinates which are obtained from the computer's star catalog.

If one of the other techniques (0, 1, or 3) is used, at least one of the directions A and B is not that of a star (or celestial body) and is determined by other means.

The sources of the four unit LOS vectors (or IMU alignment vectors) for each technique in program P-57 are given in Fig. 6.2-1. If Technique 0 is used, $\underline{s}_A$ and $\underline{s}_B$ represent the present directions of the LM Y and Z axes with respect to the IMU stable member and $\underline{s}''_A$ and $\underline{s}''_B$ represent the directions of these axes in basic reference coordinates as determined from stored information of their directions in the Moon-Fixed Coordinate System (MFCS) which were computed from the previous alignment of the IMU.

Technique 1 in Fig. 6.2-1 differs from Technique 0 in that vector $\underline{s}_A$ represents the direction of lunar gravity as determined by the Gravity Vector Determination Routine and $\underline{s}''_A$ is the direction of the landing site position vector in basic reference coordinates based upon its stored vector in moon-fixed coordinates. It is assumed that the gravity and landing site position vectors represent the same direction in space.

Technique 3 in Fig. 6.2-1 uses the gravity vector approach of Technique 1 in conjunction with one celestial body sighting with the AOT to obtain the four IMU alignment vectors. Additional information on the manner in which these vectors are computed for each technique is given later.

Alignment Technique	Sources of IMU Alignment Vectors			
	Basic Reference Coordinate System		Present IMU Stable Member Coordinate System	
	$\underline{s}''_A$	$\underline{s}''_B$	$\underline{s}_A$	$\underline{s}_B$
0	LM Y-axis stored in MFCS	LM Z-axis stored in MFCS	Present LM Y-axis	Present LM Z-axis
1	Landing Site stored in MFCS	LM Z-axis stored in MFCS	Gravity Vector Determination Routine	Present LM Z-axis
2	<u>One or more of the following:</u> 1. LGC star catalog 2. Loaded by astronaut 3. Ephemeris (i.e., SUN, EARTH)		AOT Sighting	AOT Sighting
3	Landing Site stored in MFCS	Same as Technique 2 above	Gravity Vector Determination Routine	AOT Sighting

Fig. 6.2-1 Sources of IMU **Alignment Vectors** in Program P-57

Whenever the IMU is aligned by Techniques 1, 2, or 3, one of the last functions performed by P-57 before termination is to determine and store the orientations of the LM Y and Z axes with respect to the Moon-Fixed Coordinate System. These orientations are indicated as unit vectors $\underline{u}_{YMF}$ and $\underline{u}_{ZMF}$ and completely specify the attitude of the LM (or the navigation base) with respect to the Moon Fixed Coordinate System. This procedure is generally referred to as storing the LM attitude. The LM attitude is stored primarily for backup purposes so that the IMU can be aligned at a later time using just the stored LM attitude data in case the IMU orientation is not known and there is insufficient time to perform an alignment by other means, such as in emergency launch. In addition to storing the LM attitude after each alignment with Techniques 1, 2, or 3, the LM attitude is also stored by the Landing Confirmation Program (P-68) after the vehicle has come to rest on the lunar surface. Whenever the LM attitude is stored, the Attitude flag is set to denote this fact. It should be noted that the vectors $\underline{u}_{YMF}$ and $\underline{u}_{ZMF}$ will always represent the correct orientations of the LM Y and Z axes with respect to the Moon-Fixed Coordinate System as long as there is no change in the vehicle attitude since the time when the vectors were computed.

To determine the vectors $\underline{u}_{YMF}$ and $\underline{u}_{ZMF}$, the LGC first computes the unit vectors $\underline{u}_{YREF}$ and $\underline{u}_{ZREF}$ defining the directions of the LM (or navigation base) Y and Z axes with respect to the Basic Reference Coordinate System as follows:

$$\underline{u}_{YREF} = [\text{ REFSMMAT }]^T \underline{y}_{NB}$$

$$\underline{u}_{ZREF} = [\text{ REFSMMAT }]^T \underline{z}_{NB} \qquad (6.2.1)$$

where [REFSMMAT] ^{T} is the transpose of [REFSMMAT] defined in Section 5.6.3.4 and $\underline{y}_{NB}$ and $\underline{z}_{NB}$ define the directions of the navigation base Y and Z axes with respect to the present IMU Stable Member Coordinate System and are computed by the routine CALCSMSC of Section 5.6.3.2.5. The vectors $\underline{u}_{YMF}$ and $\underline{u}_{ZMF}$ are obtained by transforming $\underline{u}_{YREF}$ and $\underline{u}_{ZREF}$, respectively, to the Moon-Fixed Coordinate System using the Planetary Inertial Orientation Subroutine of Section 5.5.2 where the time specified to this subroutine is the present time. It should be noted that it is not necessary to record the time of storage of these vectors. Whenever it is desired to use the stored vectors $\underline{u}_{YMF}$ and $\underline{u}_{ZMF}$ to obtain the orientations of the LM Y and Z axes with respect to the Basic Reference Coordinate System for a given time, it is only necessary to transform the stored vectors to the Basic Reference Coordinate System using the Planetary Inertial Orientation Subroutine with the desired time as one of the inputs.

The logic associated with the Lunar Surface Align Program (P-57) is shown in Fig. 6.2-2. At the beginning of program P-57 the astronaut indicates the desired IMU stable member orientation just as is done for the IMU Realign Program (P-52 of Section 5.6.2.1.2) except that the Nominal orientation is not used in P-57. If the Landing Site Orientation Option is selected, a time t_L must be specified by the astronaut which defines the desired IMU orientation in the manner shown in Section 5.6.3.4.4. In Fig. 6.2-2 it is seen that the initial value of t_L displayed to the astronaut is equal to the time of ignition t_{IG} for powered ascent. This value of t_L may be changed by the astronaut if he so desires. However, note that the following steps are taken in the program when the following values are loaded for t_L:

1)	$t_L = 0$	The program will set t_L equal to the present time.
2)	$t_L >$ Present Time	The program will set t_{IG} equal to t_L. This feature permits the astronaut to load a new value for t_{IG} in P-57 without having to use P-12 prior to P-57 for this purpose. It should be noted that this value of t_{IG} will be that displayed to the astronaut at the start of P-12.

After the desired IMU orientation has been selected, the astronaut indicates which of the four previously mentioned techniques he wishes to use to align the IMU. In addition, the status of the REFSMFLAG, and Attitude flags are displayed in a data code. This is followed by a check on the status of REFSMFLAG and possibly the Attitude Flag as shown in Fig. 6.2-2 to see if any previous IMU alignment information is available. The REFSMFLAG is a flag which is set whenever the computer knows the present orientation of the IMU stable member with respect to the Basic Reference Coordinate System, which also implies that the computer has the correct matrix REFSMMAT for transforming vectors from the Basic Reference Coordinate System to the present IMU Stable Member Coordinate System. Since REFSMFLAG is set only if the IMU is on and at a known orientation, its presence at the start of an alignment program generally implies that the IMU has previously been aligned and has remained in this condition except for IMU drift until

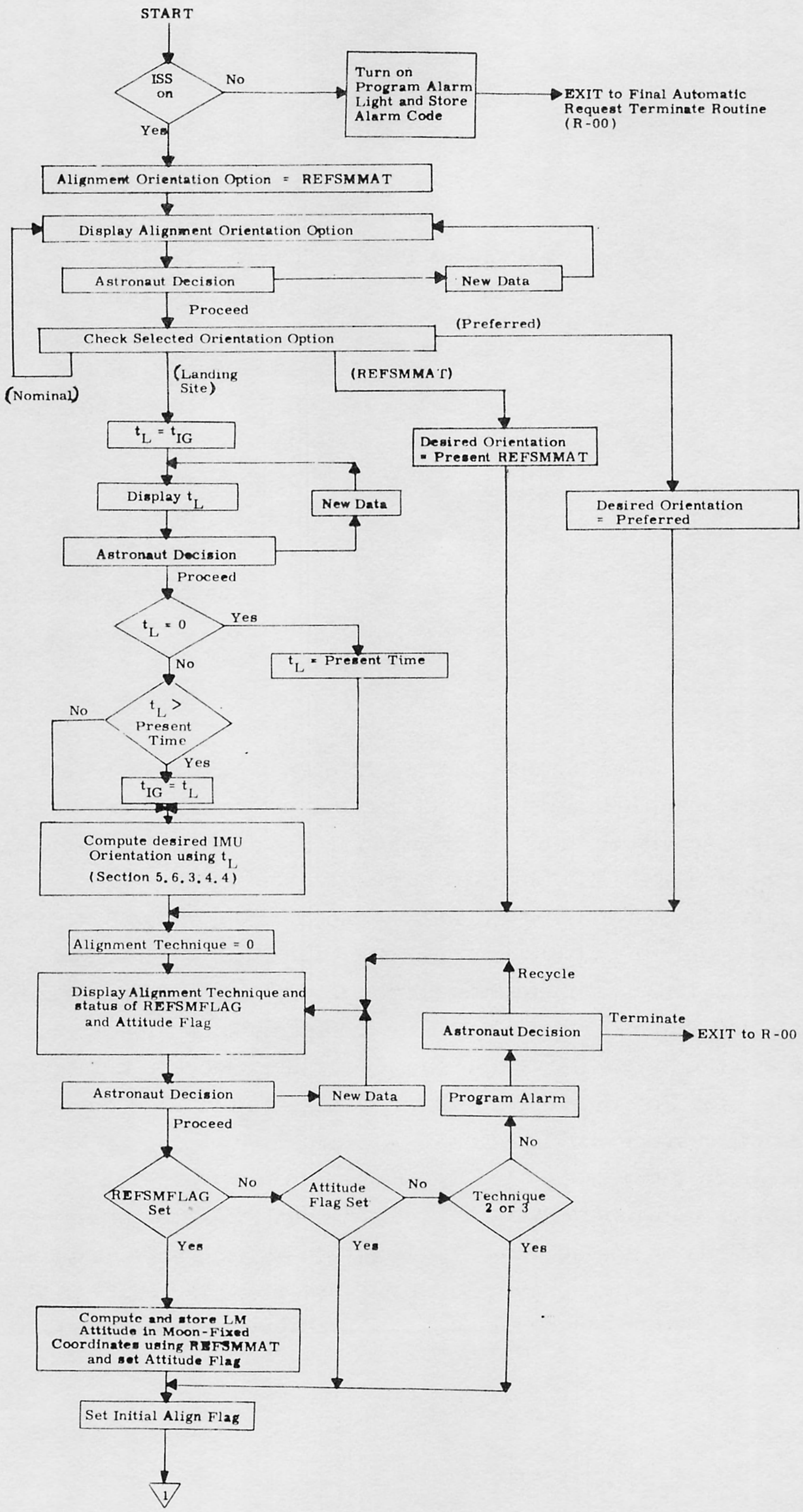


Figure 6.2-2 Lunar Surface Align Program
(page 1 of 3)

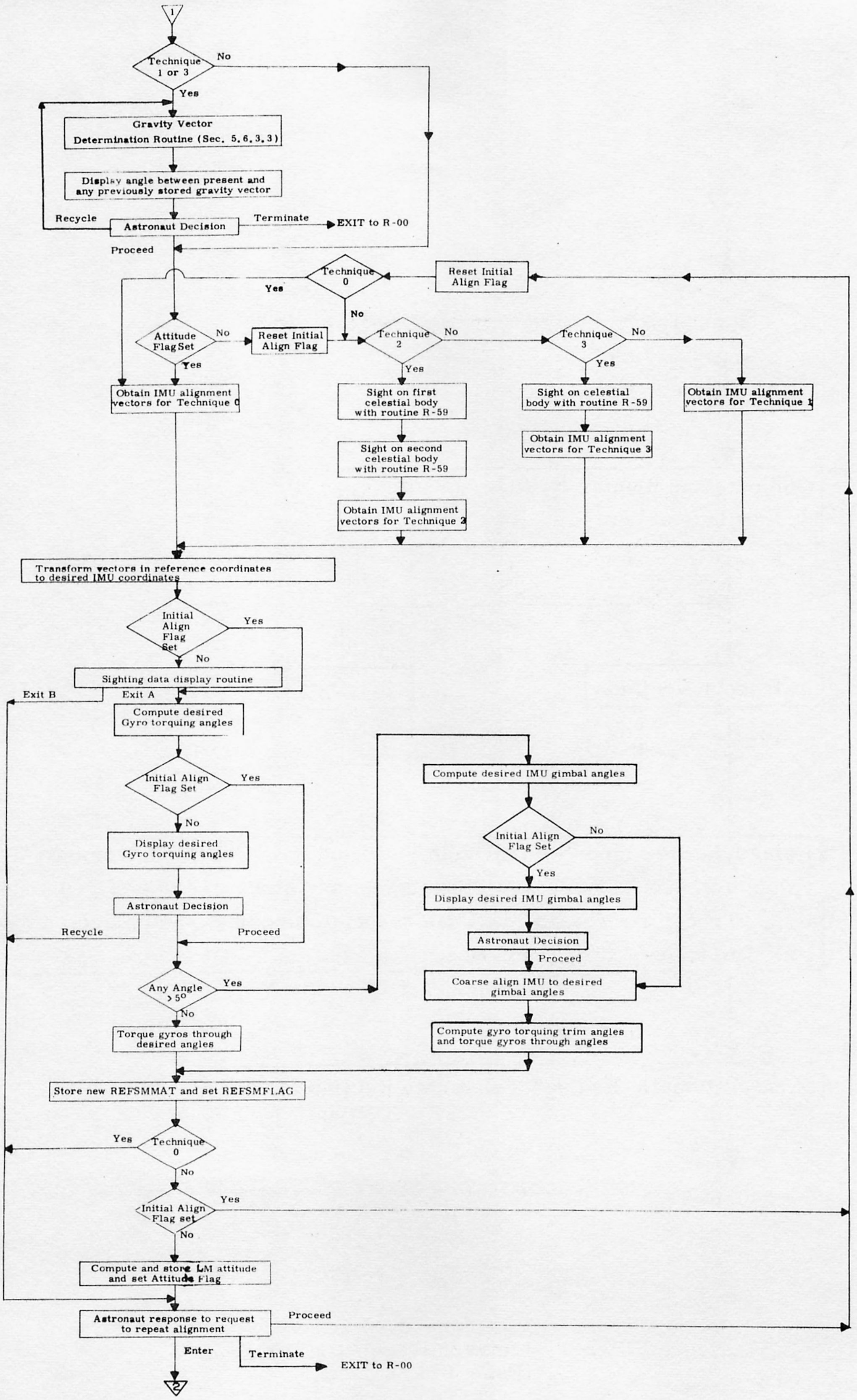


Figure 6.2-2 Lunar Surface Align Program
(Page 2 of 3)

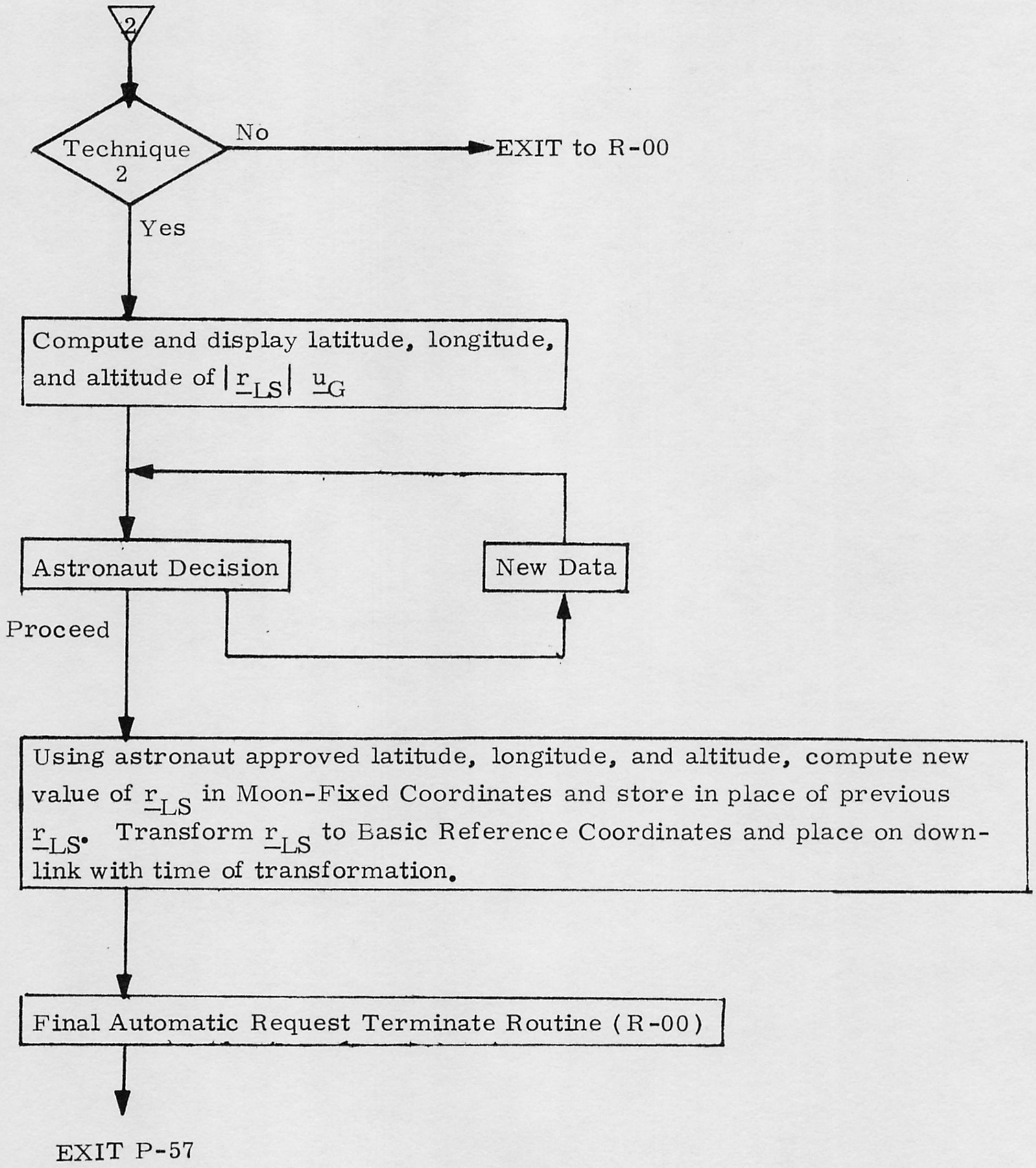


Figure 6.2-2 Lunar Surface Align Program
(Page 3 of 3)

the present alignment process. However, it is possible that the correct REFSMMAT could have been determined on earth in one of the backup situations described later and entered into the LGC without a previous alignment having been made with P-57. If the REFSMFLAG is set, the program will compute and store new LM attitude vectors based upon the present alignment of the IMU. The new attitude vectors replace any previously-stored attitude vectors and are used in the initial IMU alignment described later. It should be noted that the new LM attitude vectors and the subsequent alignment with these vectors will have the same error as present between the REFSMMAT and the IMU orientation when entering the program. If the REFSMFLAG is not set and the Attitude flag is set, the program will use the attitude vectors computed and stored in the previous use of P-57 or P-68. If the REFSMFLAG and Attitude flags are not set and Technique 0 or 1 is selected, it is seen in Fig. 6.2-2 that a program alarm is issued since these techniques require the use of stored LM attitude data.

The details associated with the alignment procedure for each technique in the remaining portion of Fig. 6.2-2 are presented in the following sections. One important feature which should be noted in Techniques 1, 2, and 3 is that an initial alignment is made to the desired IMU orientation prior to the final alignment if the Attitude flag is set. The initial alignment is made with the LM attitude data just as in Technique 0 and the final alignment is made using the regular alignment vectors for Techniques 1, 2, or 3. Since the initial alignment can be as good as the final alignment the gyro torquing angles displayed to the astronaut in the final alignment process should be near zero if an initial alignment was made. In this case, the gyro torquing angles are a direct indication of the difference between the alignment obtained with the LM attitude data and the final alignment obtained with Techniques 1, 2, or 3. If these angles are not near zero (i.e. within acceptable tolerances, such as the desired IMU alignment accuracy) this either indicates that the final alignment may not be acceptable because of bad star sightings, etc. and/or the LM attitude data is in error because of errors in the previous

alignment used to compute the LM attitude data and/ or errors due to settling of the spacecraft since the time when the LM attitude data was computed. In Fig. 6.2-2 it is seen that the astronaut can repeat the final alignment if he so desires. It should be noted that the astronaut can determine changes in the LM attitude with respect to the local gravity vector by using the Gravity Vector Determination Routine in Techniques 1 and 3.

5.6.2.2.2 Technique 0 - IMU Alignment Using the LM Attitude Data

Technique 0 is selected by the astronaut if he wants the IMU to be driven to the desired orientation using the stored LM attitude data or the present REFSMMAT. Note in Fig. 6.2-2 that if REFSMFLAG is set, the program will use the present REFSMMAT to compute a new set of LM attitude data for use in this technique. This technique provides a quick means of alignment for emergency launch in case there is insufficient time to perform the alignment by other means. If neither the Attitude flag or the REFSMFLAG is set at the beginning of this technique, it is seen in Fig. 6.2-2 that a program alarm is issued.

To obtain the four IMU alignment vectors ($\underline{s}_A$, $\underline{s}_B$, $\underline{s}''_A$, $\underline{s}''_B$) indicated in Fig. 6.2-1 for Technique 0, the following is done. The stored LM attitude vectors $\underline{u}_{YMF}$ and $\underline{u}_{ZMF}$ are transformed from moon-fixed to basic reference coordinates using the Planetary Inertial Orientation Subroutine with the present time being specified to this subroutine, and are denoted as $\underline{s}''_A$ and $\underline{s}''_B$, respectively. The unit vectors $\underline{y}_{NB}$ and $\underline{z}_{NB}$ defining the orientations of the LM Y and Z axes with respect to the present IMU Stable Member Coordinate System are computed by the routine CALCSMSC of Section 5.6.3.2.5 and are denoted as $\underline{s}_A$ and $\underline{s}_B$, respectively.

After the four IMU alignment vectors ($\underline{s}_A$, $\underline{s}_B$, $\underline{s}''_A$, and $\underline{s}''_B$) have been collected for Technique 0, it is seen in Fig. 6.2-2 that the same procedure is used to align the IMU with these vectors as is done with the alignment vectors for any other technique. The vectors $\underline{s}''_A$ and $\underline{s}''_B$ are transformed to the desired IMU Stable Member Coordinate System as follows:

$$\underline{s}'_A = [\,\text{REFSMMAT}\,]_D\, \underline{s}''_A$$

$$\underline{s}'_B = [\,\text{REFSMMAT}\,]_D\, \underline{s}''_B \qquad (6.2.2)$$

where $[\text{REFSMMAT}]_D$ denotes the desired REFSMMAT. If the Initial Align flag is set, as is the case for the first pass through Technique 0, the Sighting Data Display Routine (R-54 of Section 4) and the display of gyro torquing angles are by-passed as shown in Fig. 6.2.2. However, the program does compute the gyro torquing angles required to drive the IMU stable member from the present to the desired orientation. These angles are computed by the routine CALCGTA of Section 5.6.3.2.3 using the unit vectors $\underline{x}$, $\underline{y}$, and $\underline{z}$ defining the orientations of the desired stable member axes with respect to the present Stable Member Coordinate System, which are computed by the routine AXISGEN of Section 5.6.3.2.4 using the previously computed vectors $\underline{s}_A$, $\underline{s}_B$, $\underline{s}'_A$, and $\underline{s}'_B$. Afterwards, a magnitude check is made on each angle as shown in Fig. 6.2-2 to determine whether the IMU should be aligned by torquing the gyros through the above angles or should first be coarse aligned to the desired IMU gimbal angles and then trimmed to the desired orientation by torquing the gyros.

The desired gimbal angles used in the coarse alignment are computed by the routine CALCGA of Section 5.6.3.2.2 where the inputs to this routine are the unit vectors $\underline{x}$, $\underline{y}$, and $\underline{z}$ previously

computed by the routine AXISGEN, and the unit vectors $\underline{x}_{NB}$, $\underline{y}_{NB}$, and $\underline{z}_{NB}$ defining the orientations of the navigation base axes with respect to the present Stable Member Coordinate System, which are computed by the routine CALCSMSC of Section 5.6.3.2.5. The desired gimbal angles are displayed to the astronaut so that he can approve the resulting IMU orientation before coarse aligning to it during the initial alignment (or first pass) in P-57.

The gyro torquing trim angles are computed as follows: The unit vectors $\underline{x}_{NB}$ and $\underline{y}_{NB}$ defining the orientations of the navigation base X and Y axes with respect to the present IMU Stable Member Coordinate System are computed by the routine CALCSMSC of Section 5.6.3.2.5. The unit vectors $\underline{x}'_{NB}$ and $\underline{y}'_{NB}$ defining the orientations of the same two axes with respect to the desired IMU Stable Member Coordinate System are computed as follows:

$$\begin{aligned} \underline{x}'_{NB} &= [\,NBSM\,]_D \begin{pmatrix}1\\0\\0\end{pmatrix} \\ \underline{y}'_{NB} &= [\,NBSM\,]_D \begin{pmatrix}0\\1\\0\end{pmatrix} \end{aligned} \qquad (6.2.3)$$

where $[\,NBSM\,]_D$ is the matrix for transforming vectors from navigation base to desired stable member coordinates using the desired IMU gimbal angles computed previously for coarse alignment. Once the vectors $\underline{x}_{NB}$, $\underline{y}_{NB}$, $\underline{x}'_{NB}$, and $\underline{y}'_{NB}$ have been computed, they are used in the routine AXISGEN of Section 5.6.3.2.4 to compute the vectors $\underline{x}$, $\underline{y}$, and $\underline{z}$ defining the orientations of the desired stable member axes with respect to the present IMU Stable Member Coordinate System, which are used by the routine CALCGTA to compute the gyro torquing trim angles.

After the IMU has been aligned to the desired orientation with Technique 0, it is seen at the bottom of Fig. 6.2-2 that the program stores the new REFSMMAT and sets REFSMFLAG before requesting the astronaut to repeat the alignment with Technique 0 if he so desires. If he keys in "PROCEED", it is seen in Fig. 6.2-2 that the same

alignment will be made with Technique 0 except that the Initial Align flag will not be set, thereby causing the Sighting Data Display Routine and the display of gyro torquing angles to not be by-passed. The purpose of the Sighting Data Display Routine is to compute and display the angular difference $\phi - \phi'$ where ϕ' is the angle between $\underline{s}'_A$ and $\underline{s}_B'$ and ϕ is the angle between $\underline{s}_A$ and $\underline{s}_B$.

5.6.2.2.3 Technique 2-IMU Alignment Using Two Celestial Bodies

Technique 2 is selected by the astronaut whenever a complete alignment or realignment is to be made by sighting on two celestial bodies (usually stars) with the Alignment Optical Telescope (AOT). No program alarm is issued when selecting this technique if the Attitude flag or REFSMFLAG is not set. However, if either of these flags is set, it is seen in Fig. 6.2-2 that the IMU is initially aligned with the LM attitude data just as in Technique 0. If neither of these flags is set, the IMU remains at its present orientation and sightings are made on two celestial bodies with the Lunar Surface Sighting Mark Routine (R-59) which is described in Sections 4 and 5.6.3.1.2.

It should be noted that the astronaut can select the same celestial body codes at the beginning of routine R-59 which are described for the AOT Mark Routine (R-53) in Section 5.6.2.1.1. The same methods are used in routine R-59 to obtain a line-of-sight vector to the Sun, Earth, Moon, or a Planet in basic reference coordinates, although the Moon would not be used in this case. After the astronaut selects the celestial body, the Lunar Surface Star Acquisition Subroutine of Section 5.6.3.1.3 checks to see if the body is within the field-of-view of the AOT for any of its six viewing positions and, if successful, computes the reticle rotation angles needed to acquire the body. The AOT viewing position and the reticle rotation angles are displayed to the astronaut to aid him in acquiring the celestial body.

The four IMU alignment vectors used by Technique 2 to perform the final IMU alignment are obtained as a direct result of using routine R-59 twice to sight on two celestial bodies, where $\underline{s}_A$ and $\underline{s}_B$ are the measured directions of the two bodies in the present IMU Stable Member Coordinate System and $\underline{s}''_A$ and $\underline{s}''_B$ are the directions of the celestial bodies in basic reference

coordinates, which are either obtained from the computer's star catalog, loaded by the astronaut, or computed from ephemeris data such as for the Sun or Earth.

Once the alignment vectors have been obtained for Technique 2, it is seen in Fig. 6.2-2 that the remaining steps in the alignment process are the same as described for Technique 0 in Section 5.6.2.2.2 except that Technique 2 also computes and stores the LM attitude in moon fixed coordinates and sets the Attitude Flag. The astronaut can repeat the final alignment if he so desires. At the bottom of page 2 of Fig. 6.2-2, it is seen that the astronaut can repeat the final alignment by keying in a "PROCEED". If he wishes to establish a new landing site position vector as described in Section 5.6.2.2.6, he must key in an "ENTER" after the final alignment with Technique 2.

5.6.2.2.4 Technique 3-IMU Alignment Using the Gravity Vector and One Celestial Body

Technique 3 is selected when the complete alignment is to be made by determining the lunar gravity vector with the IMU accelerometers and sighting on one celestial body with the AOT. As will be discussed later, this technique may be selected by the astronaut if he is only interested in determining the direction of the gravity vector. No program alarm is issued by program P-57 when selecting this technique if the Attitude flag or REFSMFLAG is not set.

In Fig. 6.2-2 it is seen that Technique 3 initially uses the Gravity Vector Determination Routine of Section 5.6.3.3 to obtain a unit vector $\underline{u}_G$ defining the direction of lunar gravity in navigation base coordinates. The angle between this vector and any previously stored gravity vector is displayed to the astronaut so that he may judge whether there has been any tilt of the vehicle since the previous gravity vector was stored. The first time a gravity vector is determined, the angle between this vector and the $\underline{r}_{LS}$ (the landing site vector) previously stored by P-68 is displayed. At this point the astronaut may key in a "RECYCLE" to obtain another gravity vector or key in a "PROCEED" and continue with the alignment.

If the Attitude flag is set, it is seen in Fig. 6.2-2 that the IMU is initially aligned to the desired orientation with the LM attitude data before the celestial body sighting is made for the final alignment.

After the optical sighting has been made with the Lunar Surface Sighting Mark Routine (R-59), the computer collects the four IMU alignment vectors ($\underline{s}_A$, $\underline{s}_B$, $\underline{s}''_A$, and $\underline{s}''_B$) used by Technique 3 to perform the final alignment, where $\underline{s}_B$ and $\underline{s}''_B$ were previously obtained with routine R-59 and $\underline{s}_A$ and $\underline{s}''_A$ are computed as follows:

$$\begin{aligned} \underline{s}_A &= \left[\text{NBSM}\right] \underline{u}_G \\ \underline{s}''_A &= \text{UNIT}\left[\underline{r}_{LS}\right] \end{aligned} \qquad (6.2.4)$$

where $\underline{u}_G$ is the gravity vector stored in navigation base coordinates and $\underline{r}_{LS}$ is the landing site position vector in basic reference coordinates at the present time, which is obtained from moon-fixed coordinates with the Planetary Inertial Orientation Subroutine of Subsection 5.5.2.

Once the alignment vectors have been obtained for Technique 3, it is seen in Fig. 6.2-2 that the remaining steps in the alignment process are the same as described for Technique 0 in Section 5.6.2.2.2 except that Technique 3 also computes and stores the LM attitude in moon fixed coordinates, and sets the Attitude Flag. The astronaut can repeat the final alignment if he so desires.

It should be noted in Technique 3 that Routine R-54 will display the angular difference $\phi - \phi'$ where ϕ' is the angle between the landing site position vector $\underline{r}_{LS}$ and the catalogued star vector in desired stable member coordinates, and ϕ is the angle between the gravity vector $\underline{u}_G$ and the measured star vector in present stable member coordinates. This angular difference (except for errors in the star sighting) represents that component of the angular difference between $\underline{r}_{LS}$ and $\underline{u}_G$ in the plane defined by $\underline{r}_{LS}$ and the star vector. Repeating this process on a second star in a different direction produces another component of the angular difference, thereby enabling one to obtain an estimate of the latitude and longitude of the gravity vector which is assumed to be the true site. Further improvement in the estimate can be obtained by performing extra sightings on the same and/or other stars. A convenient way of repeating star sightings is to use Exit B of the Sighting Data Display Routine and key in a PROCEED response to the request to repeat alignment (see Fig. 6.2-2).

5.6.2.2.5 Technique 1-IMU Alignment Using the Gravity Vector and the LM Attitude Data

Technique 1 is selected by the astronaut when a complete alignment is to be made by using the lunar gravity vector and the stored LM attitude data. A program alarm is issued when selecting this technique if the attitude flag or REFSMFLAG is not set.

In Fig. 6.2-2 it is seen that Technique 1 initially uses the Gravity Vector Determination Routine just as in Technique 3 to determine the unit gravity vector $\underline{u}_G$ in navigation base coordinates. Afterwards, the IMU is aligned to the desired orientation using only the LM attitude data. The final alignment is then made with the regular IMU alignment vectors for Technique 1 where $\underline{s}_A$ and $\underline{s}_A''$ are the same as for Technique 3 and $\underline{s}_B$ and $\underline{s}_B''$ are the same as for Technique 0.

5.6.2.2.6 Determination of Landing Site Position Using the Gravity Vector and a Technique 2 Alignment

After the final IMU alignment with Technique 2, it is possible for the astronaut to have a new landing site position vector $\underline{r}_{LS}$ determined in P-57 by using the latest lunar gravity vector $\underline{u}_G$ stored in navigation base coordinates. This approach is based upon the assumption that $\underline{r}_{LS}$ is in the same direction as $\underline{u}_G$. When performing this function, it is assumed that a gravity vector was previously obtained through use of Technique 1 or 3. Since the gravity vector is stored in navigation base coordinates and is therefore affected by settling of the spacecraft, it is desirable that this vector be determined just prior to performing an alignment with Technique 2.

The logic associated with the determination of $\underline{r}_{LS}$ with the gravity vector is shown on page 3 of Fig. 6.2-2. To obtain the latitude, longitude, and altitude of the proposed $\underline{r}_{LS}$ for display to the astronaut, use is made of the Latitude-Longitude Subroutine of Section 5.5.3 where the inputs to this subroutine are the present time and the vector

$$\underline{r}'_{LS} = |\underline{r}_{LS}| [REFSMMAT]^T [NBSM] \underline{u}_G$$

where $\underline{r}'_{LS}$ and $\underline{r}_{LS}$ are respectively the proposed and present landing site position vectors. If the astronaut approves the displayed latitude, longitude, and altitude, they are used in the Latitude-Longitude Subroutine to compute the new $\underline{r}_{LS}$. In addition, this $\underline{r}_{LS}$ is transformed to the Moon-Fixed Coordinate System through use of the Planetary Inertial Orientation Subroutine of Section 5.5.2 and replaces the previously-stored $\underline{r}_{LS}$.

It should be noted in Fig. 6.2-2 that the astronaut can establish a new $\underline{r}_{LS}$ by loading his own values of latitude, longitude, and altitude instead of using those based upon the gravity vector $\underline{u}_G$ and the magnitude of the present $\underline{r}_{LS}$. This feature permits him to conveniently establish a new $\underline{r}_{LS}$ based upon data from sources such as from optical tracking of the LM by the CSM during an overhead pass, or from correlation of observed lunar terrain features with onboard lunar maps.

5.6.3 IMU ROUTINES

5.6.3.1 AOT and COAS Transformations

5.6.3.1.1 Determination of the Star Line-of-Sight During Free-Fall

To perform IMU alignment during free-fall it is necessary to determine the lines-of-sight to two separate navigation stars or known celestial bodies in stable member coordinates. As a matter of convenience all celestial bodies used for this purpose shall be referred to as stars in this section. The line-of-sight to a star is determined during free-fall by making optical sightings on the star with the Alignment Optical Telescope (AOT) or the Crew Optical Alignment Sight (COAS). These optical sightings are made by varying the LM attitude so that the star crosses the X and Y reticle lines of the AOT or the equivalent X and Y lines of the COAS. The astronaut depresses the X or Y mark button whenever the star crosses the corresponding reticle line. When the star coincides with one of the reticle lines, this defines a plane containing the star. Once the location of the star has been established in two separate planes, the line-of-sight to the star can be obtained by solving for the intersection of these two planes.

The AOT is a unity power telescope with a field-of-view (FOV) of 60 degrees and can be rotated to six distinct positions about an axis parallel to the navigation base X-axis. These positions are accurately obtained by the use of detents. The center of the FOV for each of the viewing positions is defined by the azimuth (AZ_N) and elevation (EL_N) angles shown in Fig. 6.3-1 where the subscript N denotes the viewing position being used. The approximate values of these angles for the six viewing positions are:

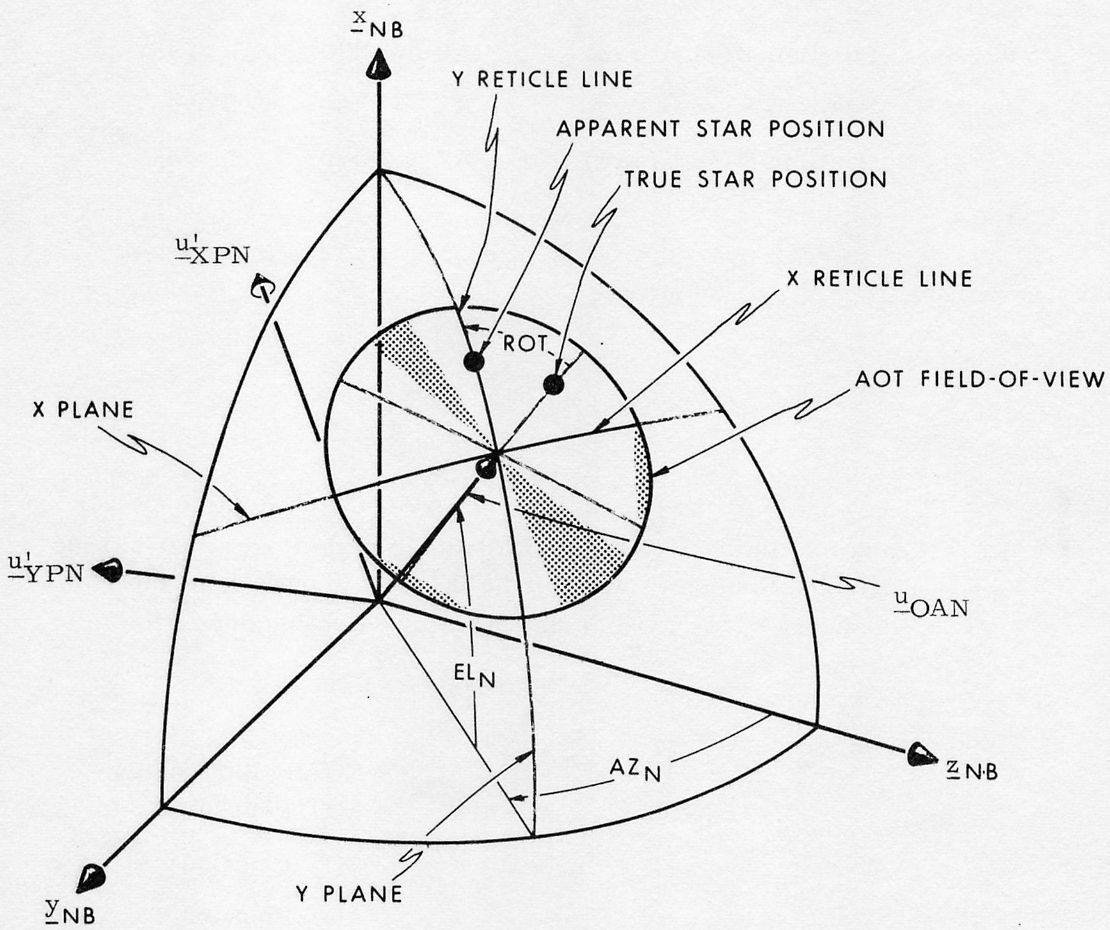


Fig. 6.3-1 AOT Alignment Geometry

Position 1 (left forward)	$AZ_1 = -60^\circ$, $EL_1 = 45^\circ$
Position 2 (forward)	$AZ_2 = 0^\circ$, $EL_2 = 45^\circ$
Position 3 (right forward)	$AZ_3 = 60^\circ$, $EL_3 = 45^\circ$
Position 4 (right rear)	$AZ_4 = 120^\circ$, $EL_4 = 45^\circ$
Position 5 (rear)	$AZ_5 = 180^\circ$, $EL_5 = 45^\circ$
Position 6 (left rear)	$AZ_6 = -120^\circ$, $EL_6 = 45^\circ$

At the beginning of the AOT Mark Routine (R-53) and the Automatic Optics (LM) Positioning Routine (R-52) the astronaut indicates whether he is going to use one of the viewing positions of the AOT or the COAS by selecting one of the following detent codes:

Detent Code	Use
0	COAS (R-52 only)
1	AOT Position 1
2	AOT Position 2
3	AOT Position 3
4	AOT Position 4
5	AOT Position 5
6	AOT Position 6
7	COAS

For detent codes 1 through 6, the computer uses the angles AZ_N and EL_N which have been pre-stored in erasable memory for the corresponding AOT viewing positions. The angles for each AOT viewing position must be stored in erasable memory since the exact values vary slightly from one AOT to another and are not known until the final AOT installation. These angles completely specify the AOT position to the computer. If detent code 7 is selected the astronaut must load-in the correct angles AZ and EL for the COAS. If the COAS is pointed along the LM+Z axis when used in this application, its AZ and EL angles are approximately zero.

Detent code 0 is only used to calibrate the COAS with Routine R-52. When this code is used by the astronaut, R-52 requests that the angles AZ and EL for the COAS be loaded just as for detent code 7. Afterward, R-52 commands the LM attitude to establish coincidence between the line of sight specified by AZ and EL and the direction of a previously-specified celestial body. The astronaut then notes any corrections that should be made in AZ and EL by observing the celestial body through the COAS. For detent code 0, R-52 will repeat its data request (i.e., detent code and COAS angles) and perform automatic optics positioning until the astronaut is satisfied with the values of AZ and EL and has indicated this by using detent code 7.

The direction of the center of the FOV (or optical axis) for each AOT viewing position and the COAS can be expressed in navigation base coordinates by the following unit vector:

$$\underline{u}_{OAN} = \begin{bmatrix} \sin(EL_N) \\ \cos(EL_N)\ \sin(AZ_N) \\ \cos(EL_N)\ \cos(AZ_N) \end{bmatrix} \qquad (6.3.1)$$

where N denotes the viewing position of the AOT (N = 1 through 6) or the COAS (N = 0 or 7).

The X and Y reticle lines of the AOT (shown in Fig. 6.3-1) and those of the COAS define two planes perpendicular to each other. The orientation of these planes with respect to the navigation base for each viewing position of the AOT and the COAS are given by the following vectors:

$$\underline{u}'_{YPN} = \begin{bmatrix} 0 \\ \cos(AZ_N) \\ -\sin(AZ_N) \end{bmatrix} \qquad (6.3.2)$$

$$\underline{u}'_{XPN} = \underline{u}'_{YPN} \times \underline{u}_{OAN}$$

where N denotes the viewing position of the AOT or the COAS (N = 7). Note that the X and Y reticle lines of the COAS are assumed to be in the planes defined by the respective vectors $\underline{u}'_{XPN}$ and $\underline{u}'_{YPN}$ in Eq. (6.3.2). Since the AOT reticle can be rotated about the AOT optical axis by means of a knob near the eyepiece, it is necessary for the astronaut to set the reticle rotation angle to zero when performing optical sightings using the X and Y reticle planes during free-fall in order for the vectors in Eq. (6.3.2) to be correct.

At this point it is necessary to consider one important difference between the AOT and the COAS which is the following. When the AOT is rotated from one viewing position to another, the apparent star field observed through the instrument rotates about the optical axis. This effect is due to the optical design of the AOT and does not occur for the AOT reticle lines. In the forward position (Position 2) the apparent star field is the same as the true star field. As the AOT is rotated through a given azimuth angle from Position 2 to 3, the apparent star field rotates with respect to the true star field by an equal amount in a clockwise direction about the AOT optical axis. For Position 3 the relationship between the apparent and true positions of a star would be as shown in Fig. 6.3-1 where the amount and sense of rotation is shown as ROT. Consequently, when an apparent star coincides with the Y reticle line, as shown in Fig. 6.3-1, the true location of this star is in a plane obtained by rotating the Y plane about the optical axis by an amount ROT in the reverse direction. If it is

assumed that the azimuth angle, AZ_N, is positive as shown in Fig. 6.3-1, then the correct sense and magnitude of the rotation (R_N) which is applied to the X and Y planes for a given AOT viewing position is

$$R_N = AZ_2 - AZ_N \quad (N = 1 \text{ thru } 6) \qquad (6.3.3)$$

Therefore, the correct orientation in navigation base coordinates of the planes represented by the AOT X and Y reticle lines is

$$\begin{aligned} \underline{u}_{XPN} &= \cos(R_N)\, \underline{u}'_{XPN} + \sin(R_N)\, \underline{u}'_{YPN} \\ \underline{u}_{YPN} &= -\sin(R_N)\, \underline{u}'_{XPN} + \cos(R_N)\, \underline{u}'_{YPN} \end{aligned} \qquad (6.3.4)$$

The orientations of the corresponding planes for the COAS are obtained by setting the rotation angle R_N to zero in Eqs. (6.3.4).

Since the above planes ($\underline{u}_{XPN}$ and $\underline{u}_{YPN}$) are fixed with respect to the Navigation Base Coordinate System it is only necessary to compute them once when using a given AOT viewing position or the COAS; regardless of the number of marks made on a star.

The orientations of the vectors $\underline{u}_{XPN}$ and $\underline{u}_{YPN}$ are obtained in stable member coordinates by

$$\underline{u}_{XP} = [NBSM]_X \, \underline{u}_{XPN}$$
$$\underline{u}_{YP} = [NBSM]_Y \, \underline{u}_{YPN} \qquad (6.3.5)$$

where $[NBSM]_X$ and $[NBSM]_Y$ are the transformation matrices based upon the IMU CDU readings stored during the X and Y marks and are defined in Section 5.6.3.2.1.

The vector describing the line-of-sight to the star in stable member coordinates is

$$\underline{s}_{SM} = \text{UNIT}\,(\underline{u}_{XP} \times \underline{u}_{YP}) \qquad (6.3.6)$$

To achieve greater accuracy in determining the line-of-sight to a star, a multiple mark capability is provided, whereby the astronaut can make up to five pairs of marks. X and Y marks may be made in any sequence convenient to the crew. The IMU CDU readings for each mark are stored until the marking process is completed. The CDU readings for X and Y marks are paired, and each pair of readings is used to obtain a star vector $\underline{s}_{SM}$ as indicated in Eqs. (6.3.5) and (6.3.6). As each star vector $\underline{s}_{SM}$ is computed it is averaged with the previously computed star vectors to obtain an average star vector $\underline{s}_n$ as follows:

$$\underline{s}_n = \frac{1}{n}\left[(n-1)\,\underline{s}_{n-1} + \underline{s}_{SM} \right] \qquad (6.3.7)$$

where n is the number of $\underline{s}_{SM}$ accumulated.

5.6.3.1.2 Determination of the Star LOS Using Lunar Surface Sighting Mark Procedure

The Lunar Surface Sighting Mark Routine (R59) may be used whenever optical sightings are made to align the IMU (with the Lunar Surface Align Program, P57; or the IMU Realign Program, P52). At the beginning of routine R-59 the astronaut selects one of the detent codes for the AOT just as described in Section 5.6.3.1.1 for the AOT Mark Routine (R-53). However, the manner in which optical sightings are made with the AOT in routine R-59 is different from that used in

routine R-53. Instead of using the entire X and Y reticle lines of the AOT, use is made of only half of the Y line and a spiral, which also exists on the AOT reticle. The complete reticle pattern is shown at the left of Fig. 6.3-2A. The spiral is so constructed as to depart radially from the center as a linear function of rotation about the center. Both the spiral and that half of the Y line used during the R59 procedure have been constructed as double lines to aid the astronaut in placing them on a star. The doubled portion of the Y line is sometimes referred to as the cursor. The entire reticle pattern can be rotated about its center by turning a knob near the eyepiece. A micrometer type readout is provided near the knob to indicate the amount of reticle rotation.

When optical sightings are made with the Lunar Surface Sighting Mark Routine both cursor and spiral measurements are required. These measurements may be made in any order convenient to the crew. A cursor measurement is made by rotating the AOT reticle pattern until the cursor line is coincident with a star and depressing a mark button to store the IMU CDU angles. The astronaut then loads the cursor angle into the LGC in response to a cursor angle load display. A spiral measurement is made by rotating the AOT reticle to bring the spiral line on the star, and depressing a mark button. The spiral angle is then loaded into the LGC. The data accumulated by the lunar surface sighting mark technique are a cursor angle (YROT) and the CDU angles corresponding to the YROT mark time, and a spiral angle (SROT) and the CDU angles corresponding to the SROT mark time. The angles are shown in Figure 6.3-2A.

The cursor mark data defines a plane containing both the sighted star and the optical axis of the AOT in platform coordinates. A spiral mark defines a spiral surface which also contains the sighted body. The line of intersection of the cursor plane $\underline{u}_{YP}$ and the spiral surface represents the direction of the star.

The normal to the cursor plane in body coordinates is computed by

$$\underline{u}''_{YPN} = -\sin(YROT)\, \underline{u}_{XPN} + \cos(YROT)\, \underline{u}_{YPN} \qquad (6.3.8)$$

where $\underline{u}_{XPN}$ and $\underline{u}_{YPN}$ are plane vectors describing the AOT X and Y reticle lines in body coordinates given by equations 6.3.4.

The cursor plane vector with respect to platform coordinates at the cursor mark time is

$$\underline{u}_{YP} = [NBSM]_C\, \underline{u}''_{YPN}$$

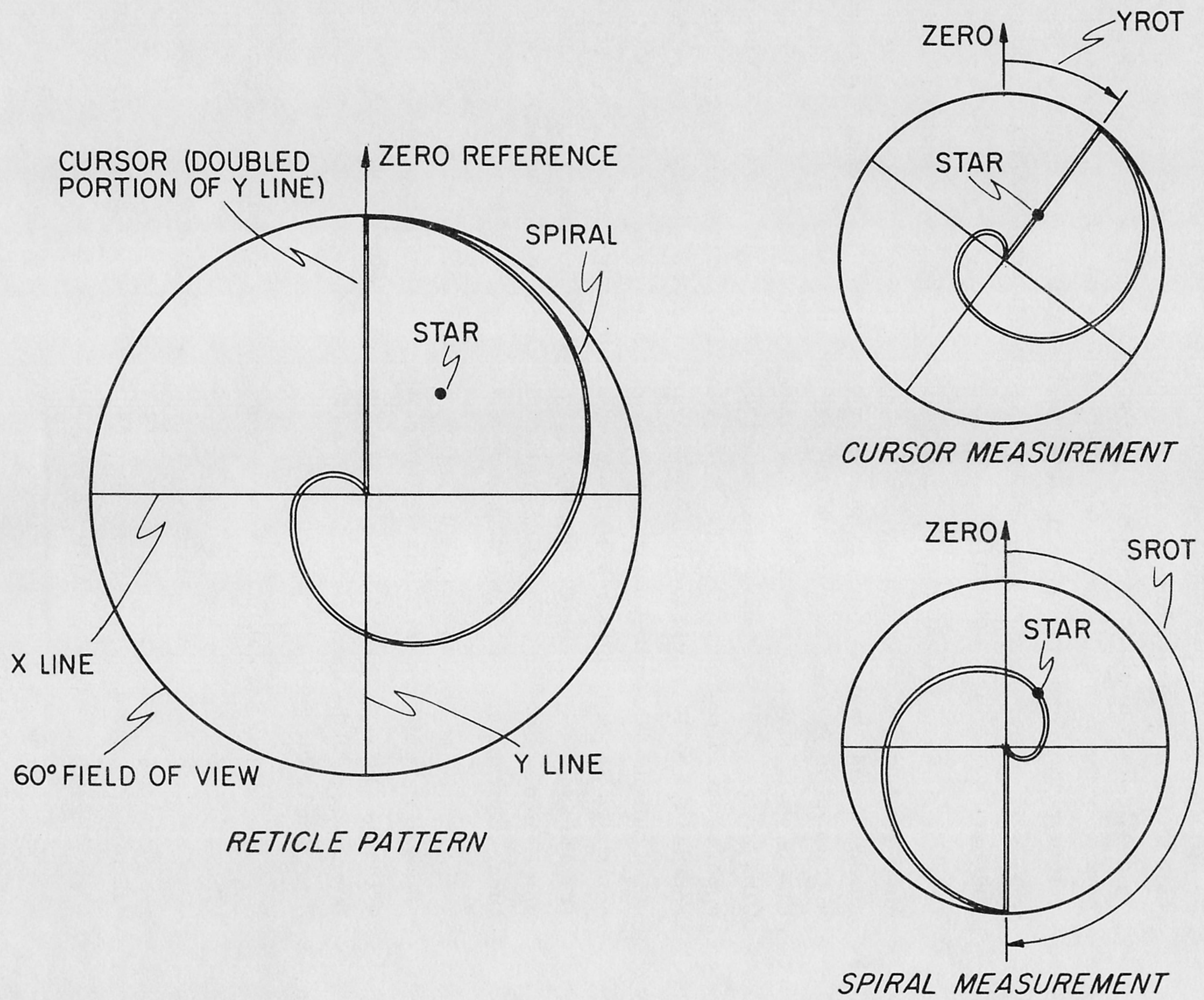


Fig. 6.3-2A AOT Reticle Pattern

$[NBSM]_C$ is the transformation matrix from body coordinates to platform coordinates using IMU CDU angles acquired by a cursor mark.

A unit vector $\underline{u}_{SP}$ is defined which sweeps along the spiral as a function of the angle θ as shown in Figure 6.3-2B. With respect to the triple primed coordinate system, $\underline{u}_{SP}$ is expressed by

$$\underline{u}_{SP} = (\cos\theta \sin\theta/12)\, \underline{u}'''_X - (\sin\theta \sin\theta/12)\, \underline{u}'''_Y + (\cos\theta/12)\, \underline{u}'''_Z \qquad (6.3.9)$$

The axis vectors $\underline{u}'''_X$, $\underline{u}'''_Y$, $\underline{u}'''_Z$ are determined with respect to the body coordinate system using the measured spiral rotation angle (SROT).

$$\underline{u}'''_Y = -\sin(\text{SROT})\, \underline{u}_{XPN} + \cos(\text{SROT})\, \underline{u}_{YPN}$$

$$\underline{u}'''_Z = \underline{u}_{OAN}$$

$$\underline{u}'''_X = \text{UNIT}(\underline{u}'''_Y \times \underline{u}'''_Z)$$

$\underline{u}_{OAN}$ is a unit vector describing the optical axis of the AOT in body coordinates given by equation 6.3.1.

During lunar surface IMU alignments the initial estimate of θ is derived from the measured rotation angles.

$$\theta = 360^\circ + \text{SROT} - \text{YROT}$$

If the surface sighting technique is used during inflight IMU alignments, θ is initially estimated using the sighting star vector stored in the LGC star catalog by transforming it to body coordinates and taking the dot product with the AOT optics axis vector.

$$\underline{\text{STAR}}_{NB} = [SMNB]_S \text{ REFSMMAT } \underline{\text{STAR}}_{REF}$$

$[SMNB]_S$ is the transformation matrix from platform to body coordinates using IMU CDU angles acquired by a spiral mark.

$$\theta = 12 \text{ ARCCOS}(\underline{\text{STAR}}_{NB} \cdot \underline{u}_{OAN}) \qquad (6.3.10)$$

To obtain the line of intersection of the cursor plane and spiral surface a vector $\underline{u}_S$ must be computed which satisfies

$$\cos\phi = \underline{u}_{YP} \cdot \underline{u}_S = 0 \qquad (6.3.11)$$

where $\underline{u}_S$ is the spiral surface vector in platform coordinates.

$$\underline{u}_S = [NBSM]_S\, \underline{u}_{SP}$$

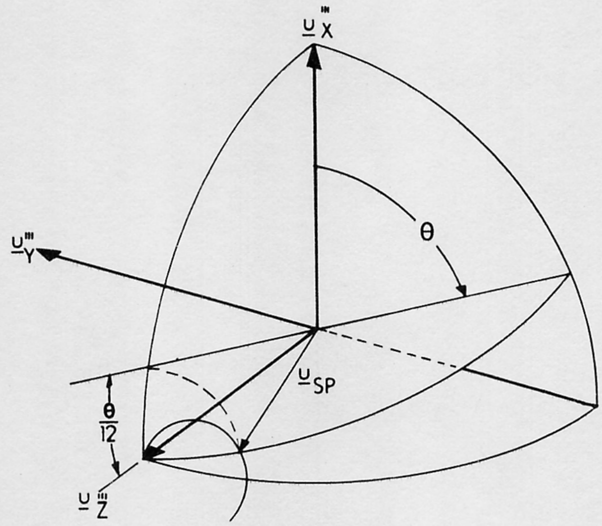


Figure 6.3-2B. Unit vector $\underline{u}_{SP}$ as a function of θ

An iteration loop is used to monitor convergence of cos ϕ where the initial computation of $\underline{u}_S$ is based on an initial estimate of θ. Subsequent computations of $\underline{u}_S$ use a θ incremented by ± 1 degree from which cos' θ is computed. The final solution for $\underline{u}_S$ represents the direction of the star in platform coordinates for a given set of mark data. The iteration loop logic is shown in Figure 6.3-2C.

To achieve greater accuracy in determining the line-of-sight to a star during the lunar stay period, a multiple mark capability is provided just as during free-fall. The star vectors $\underline{s}_{SM}$ obtained from the star sighting process are combined to obtain an average star vector $\underline{s}_n$ as described in Section 5.6.3.1.1.

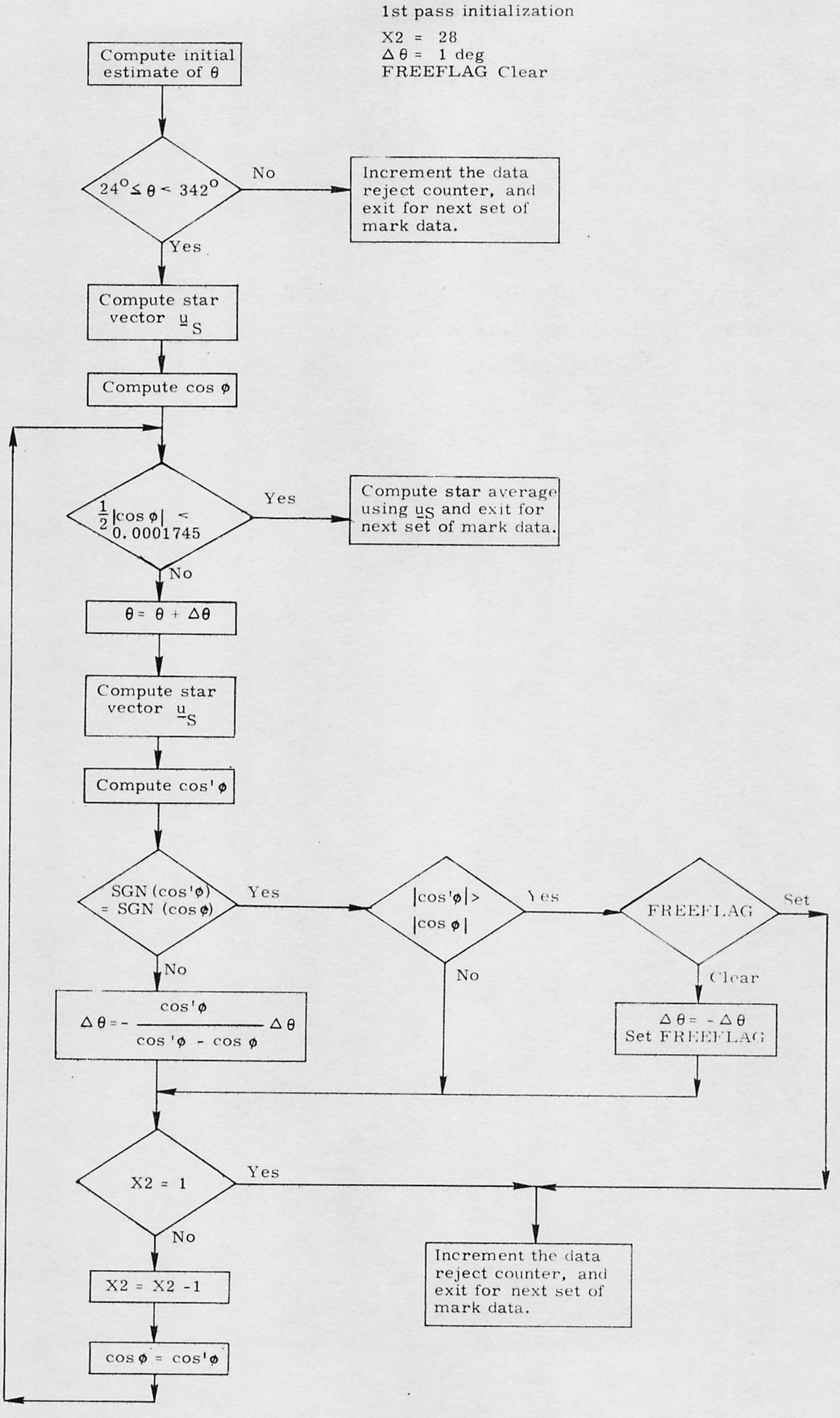


Figure 6.3-2C Iteration Loop for cos φ

5.6.3.1.3 Lunar Surface Star Acquisition Subroutine

The purpose of the Lunar Surface Star Acquisition Subroutine is to assist the astronaut in locating a navigation star with the AOT when the celestial body code (or star code) has been selected by him at the beginning of the Lunar Surface Sighting Mark Routine (R-59). The subroutine is bypassed when there is no REFSMMAT just prior to entering the subroutine.

Initially, the subroutine checks to see if the star is within 30 degrees of the center of the AOT field-of-view (FOV) for any one of the six AOT viewing positions defined in Section 5.6.3.1.1. If successful, the subroutine then computes the reticle rotation angles needed to place the cursor and spiral of the AOT reticle onto the star. The parameters computed by this subroutine for display to the astronaut are the following:

N	AOT viewing position as defined in Section 5.6.3.1.1.
YROT	Reticle rotation angle in order to place the cursor on the star for viewing position N.
SROT	Reticle rotation angle in order to place the spiral on the star for viewing position N.

If the star is not within 30 degrees of the FOV center for any AOT position, a Program Alarm is issued and the astronaut must respond as shown in Fig. 6.3-3. If the star is within 0.5 degrees of the FOV center for one of the AOT positions, the values of YROT and SROT displayed to the astronaut are zero.

The logic associated with this subroutine is shown in Fig. 6.3-3. After transforming the selected unit star vector from basic reference to navigation base coordinates, a test is made for each AOT viewing position N to see if the star vector $\underline{s}_{NB}$ is within 30 degrees of the unit vector $\underline{u}_{OAN}$ defining the optical axis or the center of the field-of-view for that viewing position. The parameters $\underline{u}_{OAN}$ and C_1 are computed for each position as follows:

$$\underline{u}_{OAN} = \begin{bmatrix} \sin 45^o \\ \cos 45^o \ \sin AZ_N \\ \cos 45^o \ \cos AZ_N \end{bmatrix}$$

(6.3.12)

$$C_1 = \underline{u}_{OAN} \cdot \underline{s}_{NB}$$

where AZ_N is the approximate azimuth angle for the AOT viewing position N which is given in Section 5.6.3.1.1.

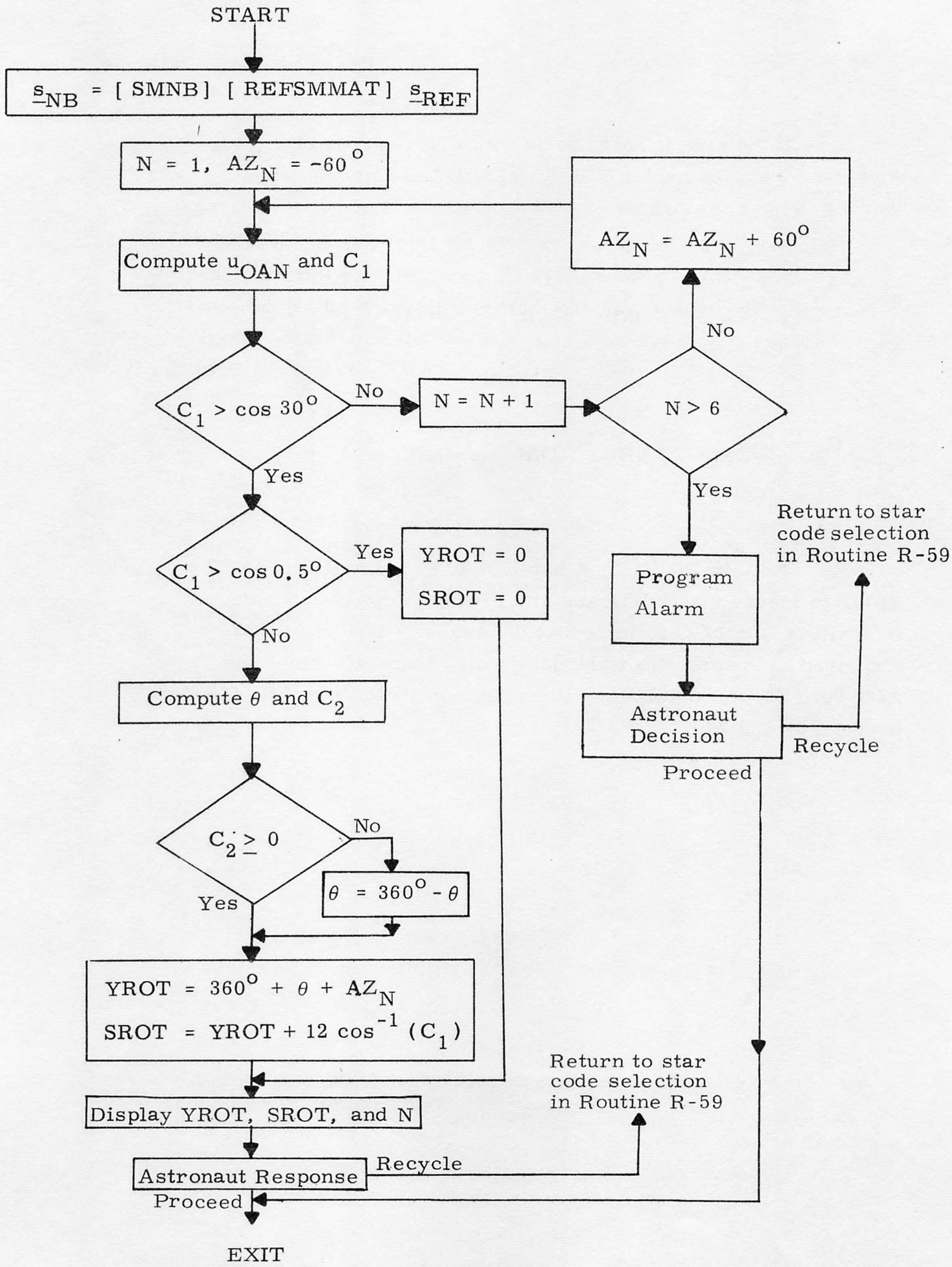


Figure 6.3-3 Lunar Surface Star Acquisition Subroutine

If the star is found to be within 30 degrees of $\underline{u}_{OAN}$, the reticle rotation angles YROT and SROT are computed as shown in Fig. 6.3-3 where:

$$\theta = \cos^{-1}\left\{ [\underline{u}_{OAN} \times \underline{s}_{NB}] \cdot [\underline{u}_{OAN} \times \begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}] \right\}$$

(6.3.13)

$$C_2 = (\underline{u}_{OAN} \times \underline{s}_{NB}) \cdot \left\{ \underline{u}_{OAN} \times [\underline{u}_{OAN} \times \begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}] \right\}$$

Note in Fig. 6.3-3 that 360° is added to the angle YROT to insure that YROT and SROT will always be positive. In addition, the LGC insures that the angles YROT and SROT displayed to the astronaut are not greater than 360° since the AOT reticle rotation dial is only capable of indicating values between 0° and 360°.

5.6.3.2 IMU Transformations

5.6.3.2.1 Stable Member-Navigation Base

Let IGA, MGA, OGA be the IMU inner, middle and outer gimbal angles, respectively. Define the following matrices:

$$Q_1 = \begin{pmatrix} \cos IGA & 0 & -\sin IGA \\ 0 & 1 & 0 \\ \sin IGA & 0 & \cos IGA \end{pmatrix} \qquad (6.3.14)$$

$$Q_2 = \begin{pmatrix} \cos MGA & \sin MGA & 0 \\ -\sin MGA & \cos MGA & 0 \\ 0 & 0 & 1 \end{pmatrix} \qquad (6.3.15)$$

$$Q_3 = \begin{pmatrix} 1 & 0 & 0 \\ 0 & \cos OGA & \sin OGA \\ 0 & -\sin OGA & \cos OGA \end{pmatrix} \qquad (6.3.16)$$

Stable Member to Navigation Base Transformation

$$\underline{u}_{NB} = Q_3 Q_2 Q_1 \underline{u}_{SM}$$

$$(6.3.17)$$

$$\left[SMNB\right] = Q_3 Q_2 Q_1$$

Navigation Base to Stable Member Transformation

$$\underline{u}_{SM} = Q_1^T \, Q_2^T \, Q_3^T \, \underline{u}_{NB}$$

$$[NBSM] = Q_1^T \, Q_2^T \, Q_3^T \qquad (6.3.18)$$

5.6.3.2.2 Calculation of Gimbal Angles (CALCGA)

Given a stable member orientation and a navigation base orientation both referred to the same coordinate system, the following procedure is used to compute the corresponding gimbal angles.

$$\underline{a}_{MG} = \mathrm{UNIT}\,(\underline{x}_{NB} \times \underline{y}_{SM})$$

$$\cos OGA = \underline{a}_{MG} \cdot \underline{z}_{NB}$$

$$\sin OGA = \underline{a}_{MG} \cdot \underline{y}_{NB}$$

$$OGA = \mathrm{ARCTRIG}\,(\sin OGA, \ \cos OGA)$$

$$\cos MGA = \underline{y}_{SM} \cdot (\underline{a}_{MG} \times \underline{x}_{NB}) \qquad (6.3.19)$$

$$\sin MGA = \underline{y}_{SM} \cdot \underline{x}_{NB}$$

$$MGA = \mathrm{ARCTRIG}\,(\sin MGA, \ \cos MGA)$$

$$\cos IGA = \underline{a}_{MG} \cdot \underline{z}_{SM}$$

$$\sin IGA = \underline{a}_{MG} \cdot \underline{x}_{SM}$$

$$IGA = \mathrm{ARCTRIG}\,(\sin IGA, \ \cos IGA)$$

where the inputs are three vectors along the stable member axes and three vectors along the navigation base axes. In the above equations ARCTRIG implies computing the angle, choosing either $\sin^{-1}$ or $\cos^{-1}$ so as to yield maximum accuracy.

5.6.3.2.3 Calculation of Gyro Torquing Angles (CALCGTA)

In the fine align procedure, after the present platform orientation is determined, the torquing angles required to move the platform into the desired orientation must be computed. This is achieved as follows:

Let $\underline{x}_D$, $\underline{y}_D$, and $\underline{z}_D$ be the desired stable member axes referred to the present stable member orientation, and let $\underline{x}_P$, $\underline{y}_P$, and $\underline{z}_P$ denote the present stable member axes. The rotations are performed in three steps: (1) rotating through θ_y about the present y axis, yielding $\underline{x}$, $\underline{y}_P$, and $\underline{z}$; (2) rotating through θ_z about the z axis, yielding $\underline{x}_D$, $\underline{y}$, $\underline{z}$; (3) and finally rotating through θ_x about the $\underline{x}_D$ axis, yielding $\underline{x}_D$, $\underline{y}_D$, $\underline{z}_D$. The relevant equations are as follows:

$$
\begin{aligned}
\underline{z} &= \text{UNIT}\,(-x_{D,3},\ 0,\ x_{D,1}) \\
\sin\theta_y &= z_1 \\
\cos\theta_y &= z_3 \\
\theta_y &= \text{ARCTRIG}\,(\sin\theta_y,\ \cos\theta_y) \\
\sin\theta_z &= x_{D,2} \\
\cos\theta_z &= z_3\ x_{D,1} - z_1\ x_{D,3} \\
\theta_z &= \text{ARCTRIG}\,(\sin\theta_z,\ \cos\theta_z) \\
\cos\theta_x &= \underline{z} \cdot \underline{z}_D \\
\sin\theta_x &= \underline{z} \cdot \underline{y}_D \\
\theta_x &= \text{ARCTRIG}\,(\sin\theta_x,\ \cos\theta_x)
\end{aligned}
\qquad (6.3.20)
$$

The required inputs are the three coordinate axes of the desired stable member orientation referred to the present stable member orientation.

5.6.3.2.4 Coordinate Axes Generator (AXISGEN)

Given two unit vectors (usually star vectors), $\underline{s}_A$ and $\underline{s}_B$, expressed in two coordinate systems, denoted by primed and unprimed characters, i.e., $\underline{s}'_A$, $\underline{s}'_B$, $\underline{s}_A$, $\underline{s}_B$, this routine computes the unit vectors $\underline{x}$, $\underline{y}$, $\underline{z}$ which are the primed coordinate system axes referred to the unprimed coordinate system. This is accomplished by defining two ortho-normal coordinate sets, one in each system, in the following manner:

$$
\begin{aligned}
\underline{u}'_X &= \underline{s}'_A \\
\underline{u}'_Y &= \text{UNIT}\,(\underline{s}'_A \times \underline{s}'_B) \\
\underline{u}'_Z &= \underline{u}'_X \times \underline{u}'_Y \\
\underline{u}_X &= \underline{s}_A \\
\underline{u}_Y &= \text{UNIT}\,(\underline{s}_A \times \underline{s}_B) \\
\underline{u}_Z &= \underline{u}_X \times \underline{u}_Y
\end{aligned}
\qquad (6.3.21)
$$

The primed coordinate system axes expressed in terms of the unprimed coordinate system axes are:

$$
\begin{aligned}
\underline{x} &= u'_{X1}\,\underline{u}_X + u'_{Y1}\,\underline{u}_Y + u'_{Z1}\,\underline{u}_Z \\
\underline{y} &= u'_{X2}\,\underline{u}_X + u'_{Y2}\,\underline{u}_Y + u'_{Z2}\,\underline{u}_Z \qquad (6.3.22) \\
\underline{z} &= u'_{X3}\,\underline{u}_X + u'_{Y3}\,\underline{u}_Y + u'_{Z3}\,\underline{u}_Z
\end{aligned}
$$

It should be noted that vectors can be transformed from the unprimed to the primed coordinate systems by using the following matrix constructed with the output (Eq. (6.3.22)) of AXISGEN:

$$
\begin{bmatrix} \underline{x}^T \\ \underline{y}^T \\ \underline{z}^T \end{bmatrix} \qquad (6.3.23)
$$

5.6.3.2.5 Calculation of Stable Member Coordinates of the Spacecraft (CALCSMSC)

To determine the directions of the X, Y, and Z axes of the present vehicle coordinate system or the navigation base coordinate system with respect to the IMU Stable Member Coordinate System, use is made of the routine CALCSMSC.

The unit vectors $\underline{x}_{NB}$, $\underline{y}_{NB}$, and $\underline{z}_{NB}$ defining the directions of the navigation base coordinate system axes with respect to the IMU Stable Member Coordinate System are determined as follows:

$$\underline{x}_{NB} = \begin{pmatrix} \cos IGA \quad \cos MGA \\ \sin MGA \\ -\sin IGA \quad \cos MGA \end{pmatrix}$$

$$(6.3.24)$$

$$\underline{z}_{NB} = \begin{pmatrix} \cos IGA \quad \sin OGA \quad \sin MGA \\ + \cos OGA \quad \sin IGA \\ - \sin OGA \quad \cos MGA \\ \cos OGA \quad \cos IGA \\ -\sin OGA \quad \sin MGA \quad \sin IGA \end{pmatrix}$$

$$\underline{y}_{NB} = \underline{z}_{NB} \times \underline{x}_{NB}$$

where IGA, MGA, and OGA are the inner, middle, and outer IMU gimbal angles, respectively. It should be noted that the rows of the transformation matrix [SMNB] also give the above vectors.

5.6.3.3 Gravity Vector Determination Routine

The Gravity Vector Determination Routine, see Fig. 6.3-4, measures the gravity vector twice, to minimize the error due to accelerometer bias and scale factor error. For the first measurement the IMU is coarse-aligned to gimbal angles of: OGA = 42°, MGA = 35.26°, IGA = -42°. In this orientation, the accelerometer (PIPA) axes are approximately at equal angles from the spacecraft X-axis, which should be roughly parallel to the gravity vector. The PIPA's are then monitored for 40 seconds, giving a gravity vector $\underline{g}$ in stable member co-ordinates. The gravity vector UNIT ($\underline{g}$) is transformed to navigation base co-ordinates, using the SMNB routine, and stored as $\underline{g}_1$.

A rotation matrix, Q, is then constructed, see Eqs. (6.3.25) and (6.3.26), which essentially defines the new orientation of the stable member axes with respect to the present stable member axes if the stable member were to be rotated 180° about the gravity vector.

$$\begin{aligned} \underline{u}'_X &= \text{UNIT}(\underline{g}) \\ \underline{u}'_Y &= \text{UNIT}\left[\underline{u}'_X \times (0,\ 1,\ 0)\right] \\ \underline{u}'_Z &= \underline{u}'_X \times \underline{u}'_Y \end{aligned} \tag{6.3.25}$$

$$Q = \begin{bmatrix} \underline{u}'_X & \underline{u}'_Y & \underline{u}'_Z \end{bmatrix} \begin{bmatrix} \underline{u}'^{T}_X \\ -\underline{u}'^{T}_Y \\ -\underline{u}'^{T}_Z \end{bmatrix} \tag{6.3.26}$$

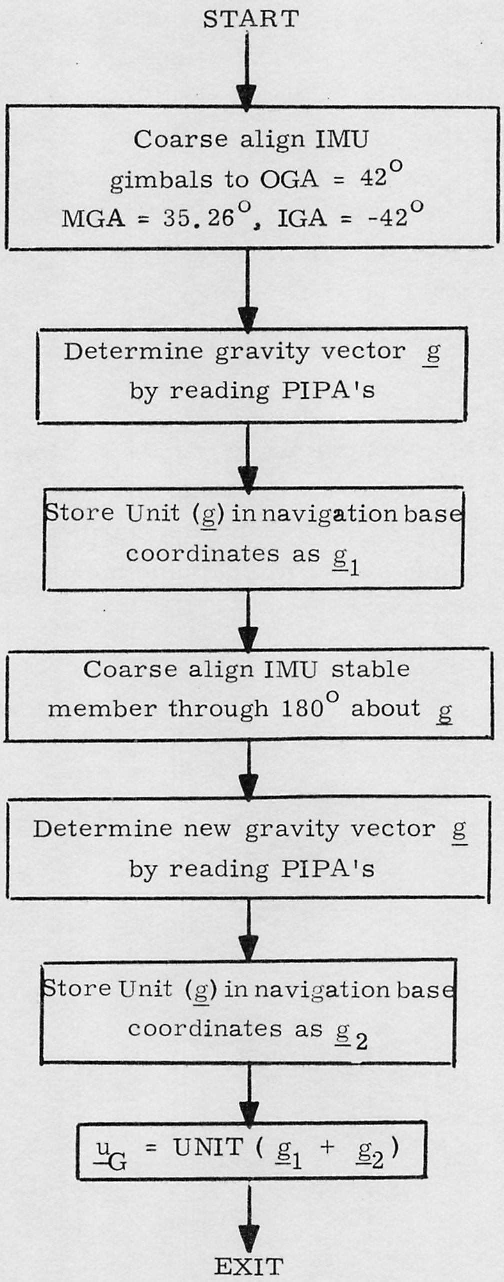


Figure 6.3-4 Gravity Vector Determination Routine

The coordinates of the desired stable member axes with respect to the present stable member axes are present in matrix Q:

$$Q = \begin{bmatrix} \underline{x}_{SM}^T \\ \underline{y}_{SM}^T \\ \underline{z}_{SM}^T \end{bmatrix} \tag{6.3.27}$$

and the coordinates of the navigation base axes ($\underline{x}_{NB}$, $\underline{y}_{NB}$, and $\underline{z}_{NB}$) with respect to the present stable member axes are computed by the routine CALCSMSC of Section 5.6.3.2.5.

With $\underline{x}_{NB}$, $\underline{y}_{NB}$, $\underline{z}_{NB}$ and the vectors in (6.3.27), the routine CALCGA can be used to determine the desired gimbal angles for the new orientation, and the PIPA's are again monitored for 40 seconds, giving another gravity vector $\underline{g}$ in stable member co-ordinates. The gravity vector UNIT ($\underline{g}$) is transformed to navigation base co-ordinates with the SMNB routine, and stored as $\underline{g}_2$.

The unit vector in the direction of the gravity vector is calculated from:

$$\underline{u}_G = \text{UNIT}\,(\underline{g}_1 + \underline{g}_2) \qquad (6.3.28)$$

By using the above procedure, the error in estimating the direction of the gravity vector because of accelerometer biases and scale factor errors is reduced.

5.6.3.4 REFSMMAT Transformations

The matrix required to transform a vector from the Basic Reference Coordinate System to the IMU Stable Member Coordinate System is referred to as REFSMMAT. This matrix can be constructed as follows with the unit vectors $\underline{u}_{XSM}$, $\underline{u}_{YSM}$, and $\underline{u}_{ZSM}$ defining the orientations of the stable member axes with respect to the Basic Reference Coordinate System:

$$\text{REFSMMAT} = \begin{bmatrix} \underline{u}_{XSM}^{T} \\ \underline{u}_{YSM}^{T} \\ \underline{u}_{ZSM}^{T} \end{bmatrix} \qquad (6.3.29)$$

5.6.3.4.1 Present REFSMMAT From Star Sightings

The present IMU stable member orientation with respect to the reference coordinate system, and the associated REFSMMAT, can be determined by sighting on two navigation stars with the AOT. If $\underline{s}'_A$ and $\underline{s}'_B$ are the unit vectors defining the measured directions of the two stars in the present stable member coordinate system, and $\underline{s}_A$ and $\underline{s}_B$ are the unit vectors to the corresponding stars as known in the reference coordinate system, then these vectors can be used as the input to the routine AXISGEN (Section 5.6.3.2.4) to obtain the present IMU orientation and REFSMMAT (Eqs. (6.3.22) and (6.3.23)).

5.6.3.4.2 Alignment for Thrusting Maneuvers (Preferred Orientation)

During certain thrusting maneuvers the IMU will be aligned according to the following equations.

$$
\begin{aligned}
\underline{u}_{XSM} &= \underline{u}_{TD} \\
\underline{u}_{YSM} &= \begin{cases} \text{UNIT}\,(\underline{u}_{XSM} \times \underline{r}) & \text{If } \underline{r} \text{ not parallel to } \underline{u}_{TD} \\ \text{UNIT}\,(\underline{u}_{XSM} \times \underline{v}) & \text{If } \underline{r} \text{ parallel to } \underline{u}_{TD} \end{cases} \\
\underline{u}_{ZSM} &= \underline{u}_{XSM} \times \underline{u}_{YSM}
\end{aligned}
\tag{6.3.30}
$$

where $\underline{u}_{TD}$ is the unit vector in the desired thrust direction at ignition and $\underline{r}$ and $\underline{v}$ are the LM position and velocity vectors.

The associated transformation matrix (REFSMMAT) is given by Eq. (6.3.29).

5.6.3.4.3 Alignment to Local Vertical in Orbit (Nominal Orientation)

The IMU stable member may be aligned to the local vertical at a specified time. For this type of orientation the stable member axes are found from the following.

$$
\begin{aligned}
\underline{u}_{XSM} &= \text{UNIT}\,(\underline{r}) \\
\underline{u}_{YSM} &= \text{UNIT}\,(\underline{v} \times \underline{r}) \\
\underline{u}_{ZSM} &= \text{UNIT}\,(\underline{u}_{XSM} \times \underline{u}_{YSM})
\end{aligned}
\tag{6.3.31}
$$

where $\underline{r}$ and $\underline{v}$ are the position and velocity vectors of the LM at the specified time, t_{align}.

The REFSMMAT associated with this IMU orientation is found from Eq. (6.3.29).

5.6.3.4.4 Lunar Landing and Launch Orientations

The proper IMU orientation for lunar landing and launch is defined by the following equations:

$$
\begin{aligned}
\underline{u}_{XSM} &= \text{UNIT}\left\{ \underline{r}_{LS} \left[t_L \right] \right\} \\
\underline{u}_{YSM} &= \underline{u}_{ZSM} \times \underline{u}_{XSM} \qquad (6.3.32) \\
\underline{u}_{ZSM} &= \text{UNIT}\left\{ \underline{h}_C \times \underline{r}_{LS} \left[t_L \right] \right\}
\end{aligned}
$$

where $\underline{u}_{XSM}$, $\underline{u}_{YSM}$, and $\underline{u}_{ZSM}$ represent the directions of the respective stable member axes expressed in the Basic Reference Coordinate System.

$\underline{h}_C$ is the orbital angular momentum vector of the CSM given by $(\underline{r}_C \times \underline{v}_C)$.

$\underline{r}_{LS}$ is the landing site position vector in the Basic Reference Coordinate System at a specified time t_L.

In Section 4 the time t_L may either be the nominal time of lunar landing referred to in IMU Realign Program (P-52) or a time specified by the astronaut at the beginning of the Lunar Surface Align Program (P-57). It should be noted that no program exists in the LGC to determine the lunar landing time, and this parameter must be supplied by RTCC.

Since the landing site moves in the Basic Reference Coordinate System because of lunar rotation, it is more convenient to store its position vector $\underline{r}_{LS}$ in the Moon-Fixed Coordinate System where it does not change with time. Whenever it is desired to express $\underline{r}_{LS}$ in the reference coordinate system for a given time (i.e., t_L) use is made of the Planetary Inertial Orientation Routine of Section 5.5.2.

The REFSMMAT associated with the landing site alignment in Eq. (6.3.32) is given by Eq. (6.3.29).

5.6.4 STAR SELECTION ROUTINE

The Star Selection Routine is used by the Inflight Fine Align Routine (R-51) to select the best pair of stars in or near the forward AOT field of view for fine alignment of the IMU. The logic diagram for this routine is shown in Fig. 6.4-1.

Each pair from the computer catalog of 37 stars is tested to see if both stars are within a 100 degree viewing cone centered with respect to the optical axis of the Alignment Optical Telescope (AOT) when in the forward viewing position (Detent 2). Although this viewing cone is larger than the actual field of view (60 degrees) of the AOT, it is used to enhance the probability that a star pair will be found with sufficient angular separation so as to insure accurate IMU alignment.

Afterwards, the routine checks to see if the angle of separation between the stars is at least 50 degrees.

If a pair passes the above tests, a check is then made to see if either star is occulted by the Sun, Earth, or Moon. The sizes of the occultation cones about each of the three bodies are such as to not only account for true occultation but to also prevent the selection of stars too near the bodies because of visibility problems. The directions and the associated occultation cone sizes of the three bodies are actually computed by the subroutine LOCSAM (Section 5.5.13) which is called by the Inflight Fine Align Routine (R-51) just prior to calling the Star Selection Routine.

The pair of stars passing the above tests and having the largest angular separation is chosen by this routine. If the routine is unable to find a satisfactory pair of stars after testing all combinations, it is seen in Routine R-51 of Section 4 that an Alarm Code is displayed, whereupon the astronaut may either repeat the star selection process at a different spacecraft attitude or select his own stars later in the Inflight Fine Align Routine (R-51).

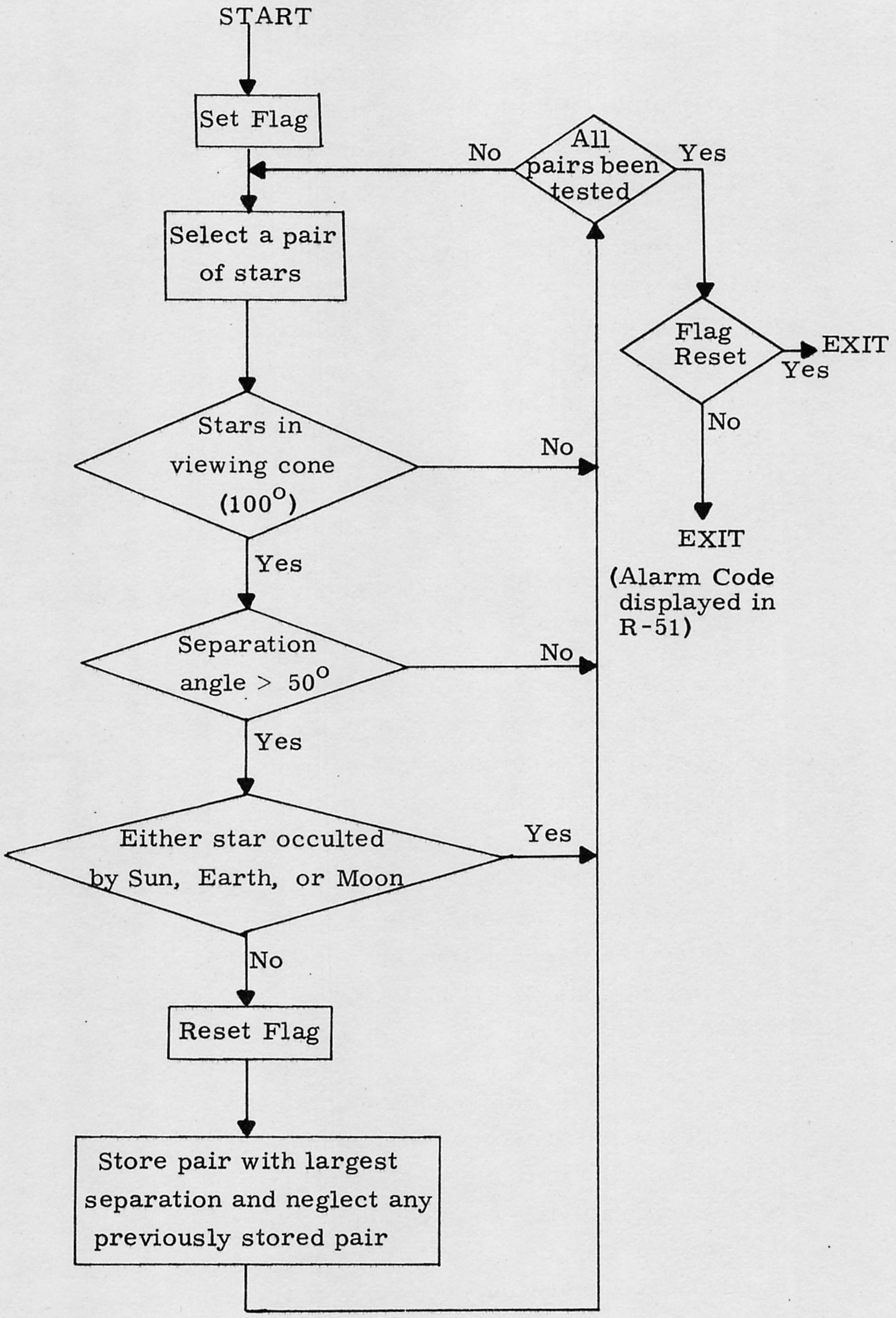


Figure 6.4-1 Star Selection Routine

5.6.5 GROUND TRACK ROUTINE

This routine is used by the astronaut in near-earth or near-moon orbit to obtain CSM or LM trajectory information. The astronaut specifies a time (GET) and a vehicle (CSM or LM). The routine uses the Coasting Integration Routine (Section 5.2.2) to extrapolate the desired vehicle's state vector to the specified time. The resulting estimated position vector is converted to latitude, longitude, altitude coordinates by means of the Latitude-Longitude Subroutine (Section 5.5.3) and these data are displayed. Altitude is defined with respect to the landing site radius for lunar orbit, and the launch pad radius for earth orbit. The astronaut can request the state vector extrapolation to continue in ten minute steps, or to another specified time, and obtain additional displays of the coordinates of points in the spacecraft's orbit.

In order to decrease the computer time required to do recycled computations of latitude, longitude, and altitude, the routine saves the results of each integration of the state as a base vector which is used as input for recycling requests.

As an additional option, the astronaut may request a display of altitude (to 10 N. M.), inertial velocity magnitude (to 1 ft/sec), and flight path angle (in degrees) at an astronaut-specified time.

5.6.6 S-BAND ANTENNA ROUTINE

The S-Band Antenna Routine (R-05) in the LGC is used to compute and display the two antenna gimbal angles which will point the antenna toward the center of the Earth, using the present LM position and body attitude. The gimbal angle definitions and their relation to the LM body axes are shown in Fig. 6.6-1.

Once the program is initiated by the astronaut the computer will automatically update the display at a rate of no greater than once per second depending on other computer activity. This update will continue until terminated by the astronaut via the DSKY. If the S-Band Antenna Routine is interrupted, the display and computation will be terminated until the astronaut reinitiates the routine.

The computational accuracy of the antenna gimbal angles is $\pm$ 1.0 degrees. This accuracy reflects only the level of accuracy of the displayed angles and does not indicate the pointing accuracy of the associated antenna alignment.

The angles computed and displayed (see Fig. 6.6-1) cover the entire range of possible gimbal angles. There is no attempt to restrict these angles to those constrained by gimbal limits or to indicate that a vehicle attitude change is required to achieve S-Band lock-on.

Rotation order α, β

α, pitch angle is rotation about the + pitch gimbal axis $(-90° < \alpha \leq 270°)$

β, yaw angle is rotation about the yaw gimbal axis fixed to the antenna $(-90° \leq \beta \leq 90°)$

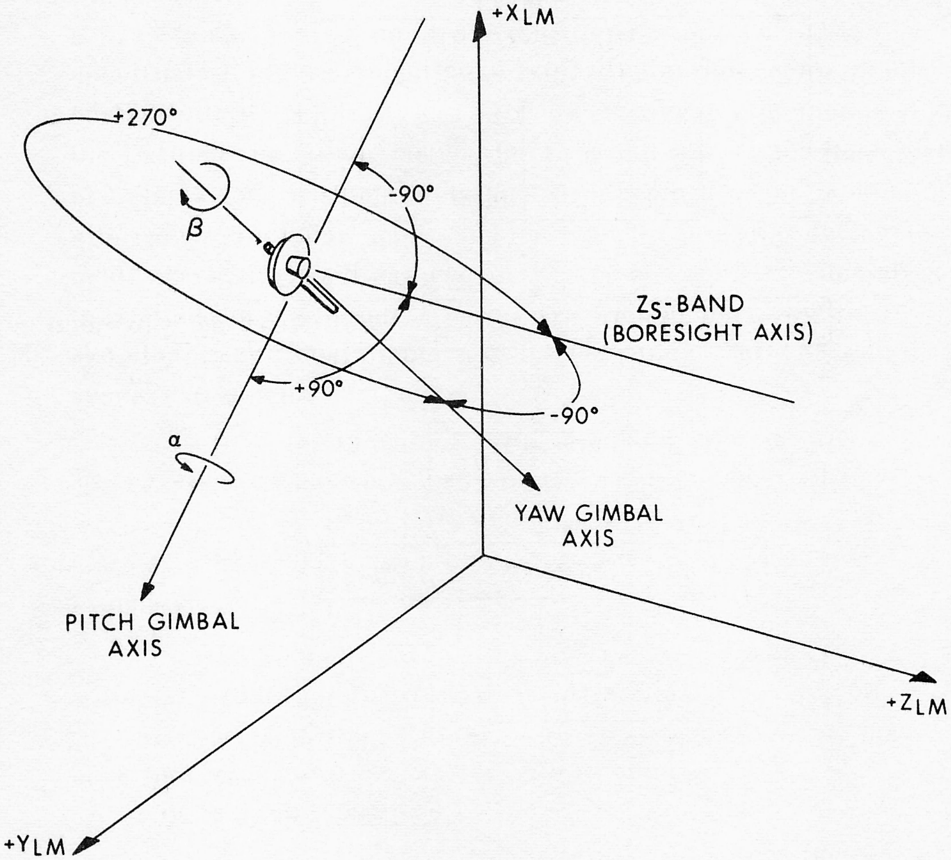


Figure 6. 6-1 Definition of LM S-Band Gimbal Angles

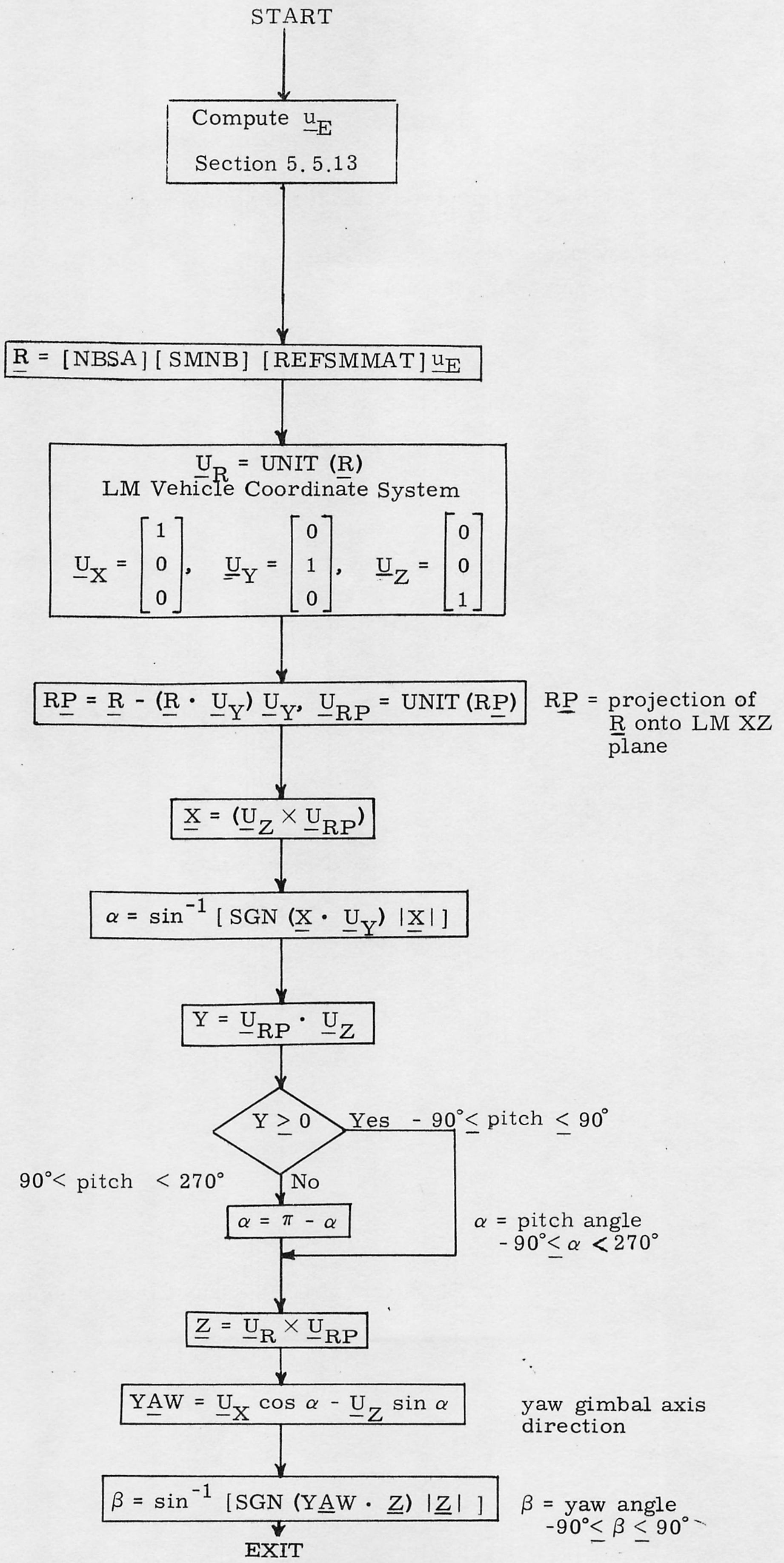


Figure 6.6-2 S-Band Angle Computations and Logic

The equations and logic used to compute the LM antenna gimbal angles are shown in Fig. 6. 6-2. The vector $\underline{R}$ from LM to Earth as determined by the unit vector $\underline{u}_E$ of the LOCSAM subroutine of Sec. 5. 5. 13 is transformed to navigation base coordinates then rotated by the transformation matrix [NBSA] to account for the orientation of the S-band antenna mount with respect to the navigation base or LM coordinates. The relation between $\underline{U}_R$ and ($U_{X_{LM}}$, $U_{Y_{LM}}$, $U_{Z_{LM}}$) determines the S-band antenna gimbal angles in the subsequent computations. The transformation matrix [NBSA] is defined as follows:

$$[NBSA] = \begin{bmatrix} \cos 45^\circ & \sin 45^\circ & 0 \\ -\sin 45^\circ & \cos 45^\circ & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

5.6.7 ADDITIONAL RENDEZVOUS DISPLAYS

During the rendezvous sequence the following routines may be called by the astronaut for the purpose of computing and displaying special quantities related to the rendezvous geometry.

5.6.7.1 Range, Range Rate, Theta Display

Routine R-31 may be called in order to display the computed range and range rate between the two vehicles and an angle θ shown in Fig. 6.7-1.

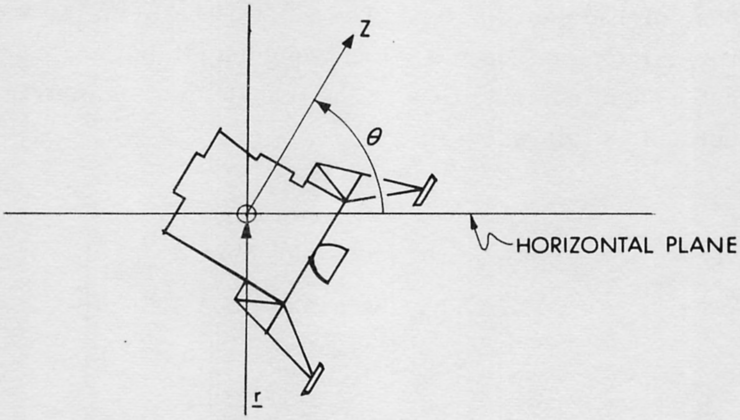


Figure 6.7-1 Definition of Theta

The angle θ represents the angle between the LM Z-body axis and the local horizontal plane of the LM. It is defined in a manner completely analogous to the definition of E in Section 5.4.2.2 and, therefore, has limits of 0 to 360 degrees.

The equations used to compute the display parameters for R-31 are given below.

$$\underline{R} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_R = \text{UNIT}(\underline{R})$$

$$\text{RANGE} = |\underline{R}| \tag{6.7.1}$$

$$\text{RANGE RATE} = (\underline{v}_C - \underline{v}_L) \cdot \underline{u}_R \tag{6.7.2}$$

To compute θ the following vector is defined

$$\underline{u}_Z = \left[\text{REFSMMAT}\right]^T \left[\text{NBSM}\right] \begin{bmatrix} 0 \\ 0 \\ 1 \end{bmatrix}$$

where NBSM and REFSMMAT are defined in Section 5.6.3 and $\underline{u}_Z$ is a unit vector along the Z-body axis expressed in basic reference coordinates. The angle θ is then found as follows:

$$\underline{u} = \text{UNIT}\,(\underline{r}_L \times \underline{v}_L)$$

$$\underline{u}_P = \text{UNIT}\left[\underline{u}_Z - \left(\frac{\underline{u}_Z \cdot \underline{r}_L}{r_L^{\,2}}\right)\underline{r}_L\right] \qquad (6.7.3)$$

$$\theta = \cos^{-1}\left[\underline{u}_Z \cdot \underline{u}_P \,\text{SGN}\,(\underline{u}_P \cdot \underline{u} \times \underline{r}_L)\right]$$

$$\text{If } \underline{u}_Z \cdot \underline{r}_L < 0;\ \theta = 2\pi - \theta$$

The three displays of R-31 are automatically updated until R-31 is terminated by the astronaut. The logic flow required to accomplish this update is shown in Fig. 6.7-2.

5.6.7.2 Final Attitude Display

Routine R-63 may be used to compute and display the FDAI angles required to point either the LM Z-axis or LM X-axis at the CSM. The choice of axis is made by the astronaut at the beginning of the routine as described in Section 4.

After initiation of this routine the state vectors of both vehicles are extrapolated to the present time plus one minute using the Conic Kepler Subroutine (Section 5.5.5). Based on these new state vectors the required gimbal angles are computed. These angles are converted to FDAI angles using the transformation described in Section 5.6.12 and the result is displayed.

There is no automatic display update; however, R-63 may easily be recycled manually.

5.6.7.3 Out-of-Plane Rendezvous Display

Routine R-36 may be used during any phase of the rendezvous sequence to provide information about the out-of-plane geometry. Three quantities (Y, $\dot{Y}$, and ψ) are computed for a given time. (The present time of ignition is automatically selected in the absence of a time determined by the astronaut.) The first two, Y and $\dot{Y}$, represent the out-of-plane position and and velocity in some sense. The third display, ψ is the angle between the

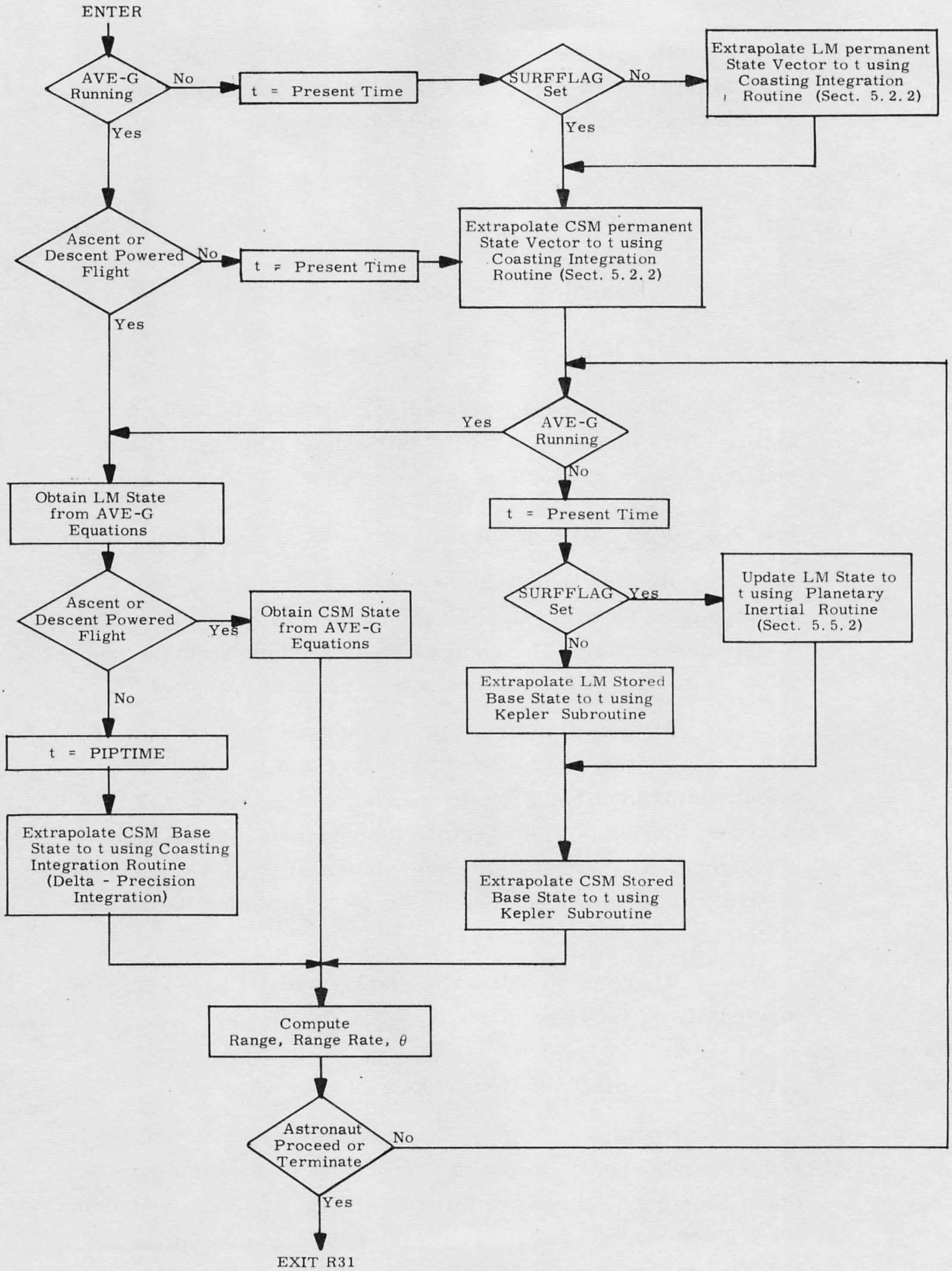


Figure 6.7-2. Range, Range Rate, θ

line of sight and the forward direction, measured in the local horizontal plane. It is equivalent to the yaw angle on the FDAI ball if the ball were aligned to an inplane, local horizontal attitude and the vehicle were rotated such that the Z-axis were pointed along the line of sight.

The exact definition of Y, $\dot{Y}$ and ψ is demonstrated by the following set of equations.

$$\underline{u} = \text{UNIT}\,(\underline{v}_C \times \underline{r}_C) \qquad (6.7.4)$$

where $\underline{r}_C$ and $\underline{v}_C$ are the position and velocity vectors of the CSM, respectively.

$$\begin{aligned} Y &= \underline{r}_L \cdot \underline{u} \\ \dot{Y} &= \underline{v}_L \cdot \underline{u} \end{aligned} \qquad (6.7.5)$$

where $\underline{r}_L$ and $\underline{v}_L$ are the position and velocity vectors of the LM, respectively.

$$\begin{aligned} \underline{u}_F &= \text{UNIT}\left[(\underline{r}_L \times \underline{v}_L) \times \underline{r}_L\right] \\ \underline{u}_R &= \text{UNIT}\,(\underline{r}_L) \qquad (6.7.6) \\ \underline{LOS} &= \underline{r}_C - \underline{r}_L \\ \underline{u}_{LOS} &= \text{UNIT}\left[\underline{LOS} - (\underline{LOS} \cdot \underline{u}_R)\,\underline{u}_R\right] \\ \psi &= \cos^{-1}(\underline{u}_{LOS} \cdot \underline{u}_F) \qquad (6.7.7) \\ \underline{N} &= \underline{u}_{LOS} \times \underline{u}_F \end{aligned}$$

$$\text{If } \underline{N} \cdot \underline{r}_L < 0;\ \psi = 2\pi - \psi$$

5.6.8 AGS INITIALIZATION ROUTINE

The Abort Guidance System (AGS) initialization involves a separate routine R-47 in the LGC that prepares a special telemetry downlink list with the required AGS initialization parameters. The AGS-LGC initialization interface is through the telemetry subsystem. At a desired GET, the astronaut may simultaneously make an "ENTER" in both LGC and AEA (Abort Electronic Assembly). The LGC stores the GET at this ENTER and subtracts this time from all state vector reference times used in the AGS initialization. There is also the option of loading an AGS initialization time obtained from an external source.

The AGS Initialization Routine next integrates the CSM and LM state vectors to the current GET. This state vector integration (Sec. 5.2.2) is based on the determination of the earth or lunar orbit condition by examining the CSM primary body indicator P_C of Section 5.2.2.6. In the earth orbital condition, the CSM and LM state vectors are divided by a factor of four and these modified state vectors are put on the special down-link telemetry list together with the difference between the state vector time and the AGS reference GET. In the lunar orbit condition the CSM and LM state vectors are modified by only the subtraction of the AGS reference GET from the state vector time before being put on the down-link list. This special telemetry down-link list is then sent 10 consecutive times before the normal down-link format is resumed.

Subsequent AGS initializations do not normally require a new AGS time referencing operation involving the simultaneous LGC and AEA ENTER inputs. During AGS initializations following the first, the AEA clock initialization may be omitted. In that case, the special initialization down-link is prepared by modifying the state vector times by subtracting the initial GET reference time determined in the first initialization.

The final operation is to zero the IMU CDU's for the AGS alignment procedure as described in Section 4 provided CDU zeroing is permissible in the LGC at that time.

5.6.9 LGC INITIALIZATION

The LGC initialization procedure prior to LM separation is a manual operation which does not involve a numbered LGC program. After the LGC is activated the first requirement is to synchronize the LGC clock with that of the CMC. This is a count-down and mark procedure described in R-33, LGC/CMC Clock Synchronization Routine of Section 4, to obtain an average clock difference which is then used to increment the LGC clock. The CMC and LGC clock synchronization can also be checked by the Mission Ground Control Center using telemetry down-link data, which can provide a more precise difference to increment the LGC clock.

Next, the following parameters are voice-linked from the CSM or uplinked from the earth to the LM to be entered into the LGC:

1)	$\underline{r}_C$	:	CSM position vector
2)	$\underline{v}_C$	:	CSM velocity vector
3)	t_C	:	CSM state vector time
4)	$\underline{r}_{LS}$	:	lunar landing site vector in moon-fixed coordinates
5)	t_0	:	time difference between zero GET and July 1.0, 1971 universal time.
6)	P_C	:	planet identifier

All of the above parameters are in octal, and all are double precision except item 5, t_0, which is triple precision and item 6, P_C, which is one bit.

The estimated CSM and LM state vectors (as defined in Eq. (2.2.28) of Section 5.2.2.6), the estimated landing site state vector, and the reference time t_0 are then initialized as follows:

$$\underline{r}_{C0} = \underline{r}_C$$

$$\underline{v}_{C0} = \underline{v}_C$$

$$\underline{r}_{Ccon} = \underline{r}_C$$

$$\underline{v}_{Ccon} = \underline{v}_C$$

$$^*\underline{\delta}_C = \underline{0}$$

$$^*\underline{\nu}_C = \underline{0}$$

$$t_C = t_C$$

$$^*\tau_C = 0$$

$$^*x_C = 0$$

$$P_C = \begin{cases} 0 \text{ for earth orbit} \\ 1 \text{ for lunar orbit} \end{cases}$$

$$\underline{r}_{L0} = \underline{r}_C$$

$$\underline{v}_{L0} = \underline{v}_C$$

$$\underline{r}_{Lcon} = \underline{r}_C$$

$$\underline{v}_{Lcon} = \underline{v}_C$$

$$^*\underline{\delta}_L = \underline{0}$$

$$
\begin{aligned}
{}^{*}\underline{\nu}_L &= \underline{0} \\
t_L &= t_C \\
{}^{*}\tau_L &= 0 \\
{}^{*}x_L &= 0 \\
P_L &= \begin{cases} 0 \text{ for earth orbit} \\ 1 \text{ for lunar orbit} \end{cases} \\
{}^{*}\underline{r}_{LS} &= \underline{r}_{LS} \\
t_0 &= t_0
\end{aligned}
$$

where the items on the left-hand side of the above equations are LGC variables, and the items on the right-hand side are the uplinked or voice-linked parameters. The subscripts C and L refer to CSM and LM, respectively. The variables marked with an asterisk (*) are initialized during the prelaunch erasable load and do not have to be reset unless the values have been changed during the mission.

5.6.11 LGC IDLING PROGRAM

This program is used to maintain the LGC in a state of readiness for entry into any other program. While the idling program is in operation, the Coasting Integration Routine (Section 5.2.2) is used to advance the estimated CSM state vector (and the estimated LM state vector when the LM is not on the surface of the moon) to approximately current time. This procedure has the lowest priority of all programs, and is performed only when no other program is active. This periodic state vector extrapolation is not necessary from a theoretical point of view, but does have two practical purposes. First, it is advisable to maintain current (or at least nearly current) state vector estimates in case an emergency situation arises. Second, a significant amount of computation time is transferred from a period of high computer activity (navigation measurement processing, targeting, etc.) to a period of low activity.

This periodic state vector extrapolation is valid for near-earth or near-moon orbits only since the capability for accurate cislunar-midcourse integration does not exist in the LGC.

In order to use the Coasting Integration Routine in an efficient manner, the maximum value for the integration time step, Δt_{max}, is computed as described in Section 5.2.2.5. Let t_C be the time associated with the estimated CSM state vector and t_1 be the current time. The estimated CSM state vector is extrapolated ahead when

$$t_1 > t_C + 4\,\Delta t_{max} \qquad (6.11.1)$$

The integration is terminated when Δt_{max} is more than the integration time-to-go. In this manner no extra and smaller-than-maximum integration time steps are performed, and the periodic integration is accomplished most efficiently.

The estimated LM state vector (if applicable) is then extrapolated to the CSM state vector time.

The error transition matrix W (See Section 5.2.2.4) is extrapolated with the estimated LM or CSM state vector if RENDWFLG indicates that the W matrix is valid. RENDWFLG is defined in Sections 5.2.4.2.2 and 5.2.5.4. The selection of which state vector to extrapolate the W matrix with depends upon whether the LM is in flight or on the moon.

The logic for the periodic state vector extrapolation is illustrated in Fig. 6.11-1. The variables D and V are indicators which control the Coasting Integration Routine. The quantities $\underline{x}_L$ and $\underline{x}_C$ are the estimated LM and CSM state vectors, respectively, and $\underline{x}$ is a temporary state vector used for integration. Refer to Section 5.2.2.6 for precise definitions of these items. The switch SURFFLAG indicates whether or not the LM is on the surface of the moon. This flag is set to one (zero) when the LM has landed on (lifted off from) the lunar surface.

As shown in the figure, time synchronization of the two state vectors is achieved and maintained by this program. The purpose of the state vector synchronization is to guarantee correct W matrix extrapolation during rendezvous navigation.

In order to permit correction of wrong erasable memory parameters which have caused or could cause an invalid and excessively lengthy integration process to begin, there is an emergency special DSKY verb to terminate or inhibit the Coasting Integration Routine. This special verb causes the following to occur:

1. If the Coasting Integration Routine is in operation, it is terminated at the end of the current time step.
2. The current program is terminated.
3. The LGC Idling Program (P-00) is activated.
4. The P-00 state vector test is bypassed so that no state vector integration test occurs until reselection of P-00.

Note that this operation does not maintain state vector synchronization and can, therefore, cause incorrect W matrix extrapolation in rendezvous navigation.

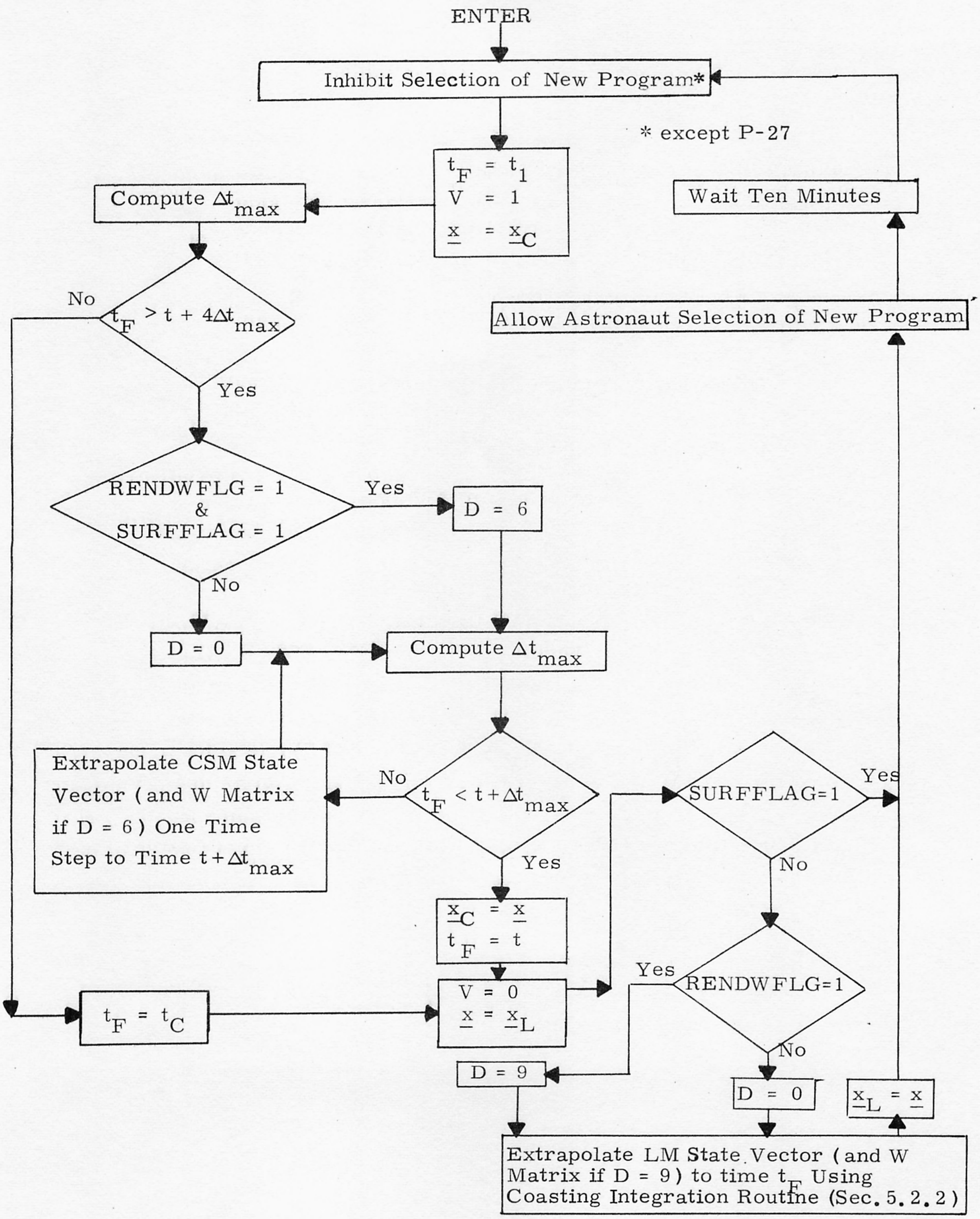


Figure 6.11-1 LGC Idling Program State Vector Extrapolation Logic Diagram

5.6.12 FDAI-IMU TRANSFORMATIONS

The following transformations are used to convert IMU gimbal angles to angular readings on the LM FDAI Ball.

YAW

$$Y = \sin^{-1}\left[-\cos(MGA)\sin(OGA)\right]$$

where the display is such that

$$90^{o} \geq Y \geq 0^{o}$$

or

$$360^{o} > Y \geq 270^{o}$$

PITCH

$$(\text{If } |\sin Y| \neq 1.0)$$

$$P = \sin^{-1}\left[\frac{\sin(IGA)\cos(OGA) + \cos(IGA)\sin(MGA)\sin(OGA)}{\cos Y}\right]$$

$$= \cos^{-1}\left[\frac{\cos(IGA)\cos(OGA) - \sin(IGA)\sin(MGA)\sin(OGA)}{\cos Y}\right]$$

ROLL

(If $|\sin Y| \neq 1.0$)

$$R = \sin^{-1}\left[\frac{\sin(MGA)}{\cos Y}\right] = \cos^{-1}\left[\frac{\cos(MGA)\cos(OGA)}{\cos Y}\right]$$

where the displays for both P and R are always positive angles less than 360°.

For $Y = 90^{\circ}$ or 270° the preceding equations are indeterminate and the pitch and roll readings become meaningless since they are not uniquely defined in terms of IMU gimbal angles because the FDAI outer and inner gimbal angles are likewise not unique. However, the following equations give the relationships between the IMU gimbal angles and the sum or differences of P and R.

If

1. $Y = 90^{\circ}$, then either

   a. $MGA = 0^{\circ}$ and $OGA = -\frac{\pi}{2}$ whence
   $P + R = IGA$

or

   b. $MGA = \pi$ and $OGA = +\frac{\pi}{2}$ whence $P + R = IGA + 180^{\circ}$

2. $Y = 270^{\circ}$, then either

   a. $MGA = 0^{\circ}$ and $OGA = +\frac{\pi}{2}$ whence $P - R = IGA$

or

   b. $MGA = \pi$ and $OGA = -\frac{\pi}{2}$ whence $P - R = IGA + 180^{\circ}$

where

IGA = inner IMU gimbal angle

MGA = middle IMU gimbal angle

OGA = outer IMU gimbal angle

The IMU gimbal angles are defined to be zero when the axes of rotation of the gimbals are mutually orthogonal. Each gimbal angle is defined as that angle through which the designated gimbal must be rotated, in the conventional right hand sense, with respect to its outer neighbor to make the X, Y, Z Coordinate Systems of both gimbals coincident.

5.6.13 IMU COMPENSATION

The IMU Compensation is designed to compensate for PIPA bias and scale factor error and at the same time accumulate gyro torquing commands necessary to compensate for the associated bias and acceleration caused gyro drifts. The correction to the PIPA's is

$$PIPA_C = (1 + SFE_I)\, PIPA_I - BIAS_I\, \Delta t$$

where

$PIPA_C$ is the compensated data for the I^{th} PIPA denoted $PIPAX_C$, $PIPAY_C$, $PIPAZ_C$

$$SFE = \frac{SF - SF_{nom}}{SF_{nom}} \quad \text{(erasable load)}^*$$

$$SF = \text{Scale-factor } \frac{CM/Sec}{Pulse}$$

$BIAS_I$ is the bias for the I^{th} PIPA (an erasable load)

The compensated data is then used to compute the IRIG torquing necessary to cancel the NBD, ADIA, and ADSRA gyro coefficients. The computations are

$$XIRIG = -ADIAX\ PIPAX_C + ADSRAX\ PIPAY_C - NBDX\ \Delta t$$

$$YIRIG = -ADIAY\ PIPAY_C + ADSRAY\ PIPAZ_C - NBDY\ \Delta t$$

$$ZIRIG = -ADIAZ\ PIPAZ_C - ADSRAZ\ PIPAY_C + NBDZ\ \Delta t$$

---

*The term "erasable load" refers to data entered in LGC erasable memory just prior to launch.

where

XIRIG, YIRIG, ZIRIG are gyro drift compensations

NBDX, NBDY, NBDZ are gyro bias drifts (an erasable load)

ADSRAX, ADSRAY, ADSRAZ are gyro drifts due to acceleration in spin reference axis (an erasable load)

ADIAX, ADIAY, ADIAZ are gyro drifts due to acceleration in the input axis (an erasable load)

When the magnitude of any IRIG command exceeds two pulses, the commands are sent to the gyros.

During free-fall only the NBDX, NBDY, NBDZ are the relevant coefficients and the routine is so ordered that only these terms are calculated for the gyro compensation.

5.6.14 RR / LR SELF TEST ROUTINE

The purpose of the RR / LR Self Test Routine (R-04 in Section 4) is to provide suitable DSKY displays and LGC down-link information to support the self tests of the Rendezvous Radar (RR) and the Landing Radar (LR). In doing this the routine also provides a check on the data transmission interface between the LGC and the two radar systems.

The self test of either radar system is accomplished by the astronaut through use of a radar test switch at the LM console. When this switch is in the RR or LR position, artificial target signals are generated within the associated radar system. These signals when acquired by the tracking networks of the radar correspond to certain known values in the measurement parameters (range, etc).

The self tests of the RR and LR can be conducted with or without the assistance of the LGC. Since these tests can interfere with the normal RR and LR data read and update functions of the LGC, it is essential that the astronaut not have the radar test switch in one of its test positions (RR or LR) when either radar is being used for update purposes.

If the astronaut wishes to have the radar self test parameters displayed on the DSKY and placed on the LGC down-link, he calls the RR / LR Self Test Routine. At the beginning of the routine he indicates which radar is being self tested. If he selects the LR, the routine will start reading continuously the four LR data parameters (v_X, v_Y, v_Z, and range), once per second. Each parameter is sampled by a single 80.001 millisecond gate. These are placed on the LGC down-link and are displayed to him in two separate DSKY displays.

One display contains the range and the present position of the LR antenna. The other display contains the three LR velocity components (v_X, v_Y, and v_Z). The LGC will transfer, display, and downlink the contents of the LR output register regardless of the status of the Range and Velocity Data Good discretes. In the absence of either of these discretes, the corresponding parameter indication may be zero because of the implementation of the LR which inhibits filling the LR output register in the absence of the particular Data Good discrete. However, whenever a given parameter is obtained from the LR, the associated data good discrete is checked and the associated DSKY light (LR Altitude Fail or LR Velocity Fail) is turned on if the discrete is missing.

If the RR test mode is selected, a check is made to see if the RR Auto Mode discrete is being received from the RR. If it is present, the RR Track Enable discrete is issued to allow the radar to lock-on the self test signals and subsequently generate the RR Data Good discrete during the data reading process. The purpose of checking the RR Auto Mode discrete is to insure that the mode control switch of the RR is in the LGC position during the data read operation. This insures that the phase of the RR resolver excitation is within tolerance for normal angle measurements by the RR CDU's. If the RR mode control switch is in either of its other two positions, this phase tolerance is not insured. If the RR Auto Mode is absent, a request is issued to the astronaut to place the RR mode control switch in the LGC position. Once this has been accomplished, the routine starts reading the RR CDU angles (shaft and trunnion) every second and the range and range rate on alternate seconds. These quantities are displayed on the DSKY in two separate displays, and thus normally will be down-linked. One display contains the RR CDU angles. It should be noted that the polarity of the displayed RR trunnion angle is opposite that defined in Fig. 6.15-1. The other display contains the range and range rate. The four RR data parameters are displayed and placed on the down-link regardless of the status of the RR Data Good discrete. However, whenever a range or range rate measurement is obtained from the RR, the RR Data Good discrete is checked and the Tracker Fail Alarm is turned on if the discrete is missing. Each range and range rate parameter obtained by this routine is the result of a single 80 millisecond gating interval.

It should be noted that the RR or LR data initially displayed by this routine may not be meaningful since the radar tracking loops may still be in the process of locking onto the internally generated target signals of the radar. In addition, the CDU data displayed at the beginning of the RR self test may be that associated with the process of zeroing the RR CDU's. Whenever the RR Auto Mode discrete is first detected by the RR Monitor Routine (R-25) of Section 5. 2. 4. 3, the RR CDU's are zeroed.

5.6.15 RR ANGLE TRANSFORMATIONS

5.6.15.1 Determination of RR Antenna Direction in Navigation Base Coordinates

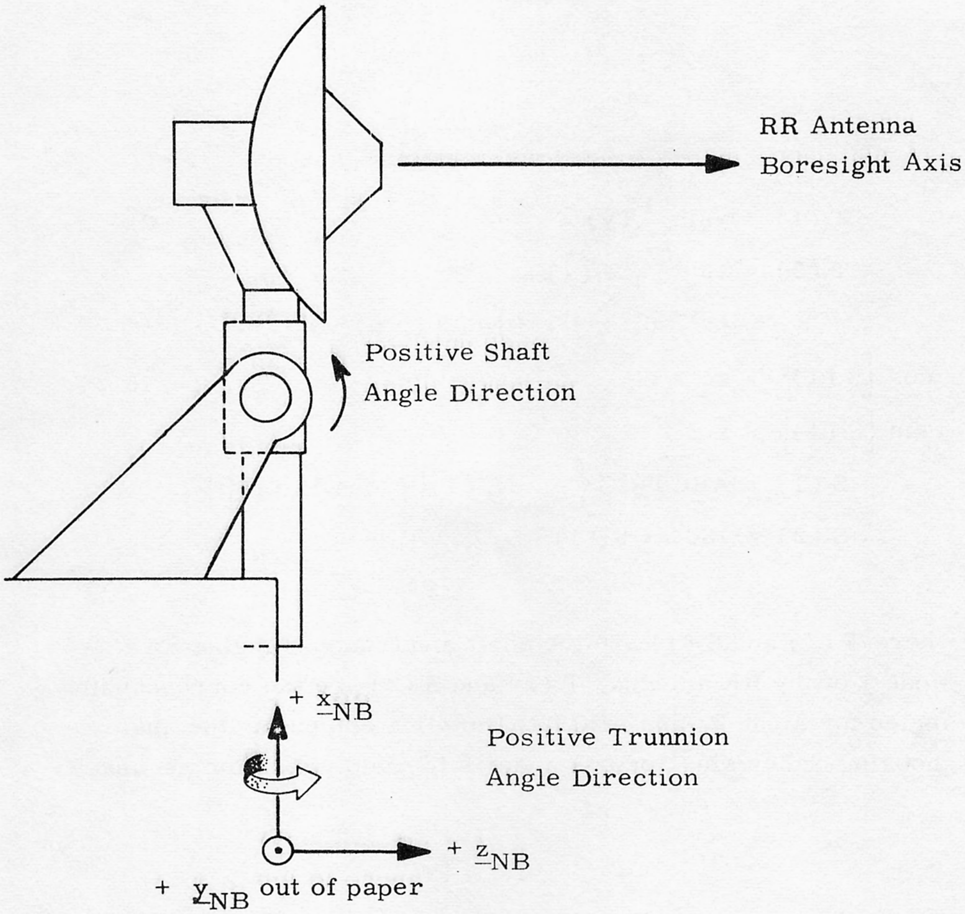
To obtain a unit vector $\underline{u}_{RR}$ specifying the present direction of the boresight axis of the RR antenna in navigation base coordinates use is made of the following:

$$\underline{u}_{RR} = \begin{bmatrix} \sin S \ \cos T \\ -\sin T \\ \cos S \ \cos T \end{bmatrix} \qquad (6.15.1)$$

where S and T are the shaft and trunnion angles of the RR antenna as indicated by the RR CDU's. The positive sense of these angles is shown in Fig. 6.15-1.

5.6.15.2 Equivalent RR Angles for a Desired Pointing Direction

Let $\underline{u}_D$ be a unit vector defining a desired pointing direction in navigation base coordinates. This direction may be a desired direction for RR designation for which the corresponding RR shaft and trunnion angles must be computed in order to see if they are within the angular coverage modes of the antenna. The corresponding shaft and trunnion angles for this direction are determined as follows:



1. $+\underline{x}_{NB}$, $+\underline{y}_{NB}$, and $+\underline{z}_{NB}$ are positive axes of Navigation Base Coordinate System.
2. Zero shaft and trunnion angles place the RR electrical boresight axis parallel to $+\underline{z}_{NB}$.
3. The antenna shaft axis is the outer gimbal axis and is parallel to $+\underline{y}_{NB}$.
4. The antenna trunnion axis is the inner gimbal axis and is orthogonal to the shaft axis and the RR electrical boresight axis.
5. Positive shaft rotation is clockwise motion of antenna as viewed along $+\underline{y}_{NB}$.
6. Positive trunnion rotation is clockwise motion of antenna as viewed along $+\underline{x}_{NB}$ when shaft angle is zero.

Figure 6.15-1 RR Shaft and Trunnion Angles

$$
\begin{aligned}
\underline{u}_D &= (x,\ y,\ z) \\
T(1) &= -\sin^{-1}(y) \\
T(2) &= 180^{\circ} - T(1) \\
\underline{u}_P &= (x',\ 0,\ z') = \text{UNIT}(x,\ 0,\ z) \\
\cos\left[S(1)\right] &= z' \qquad\qquad (6.15.2) \\
\sin\left[S(1)\right] &= x' \\
S(1) &= \text{ARCTRIG}\left\{\sin\left[S(1)\right],\ \cos\left[S(1)\right]\right\} \\
S(2) &= 180^{\circ} + S(1)
\end{aligned}
$$

where T (1) and S (1) are the shaft and trunnion angles for Mode 1 of the RR antenna, T (2) and S (2) are the corresponding angles for Mode 2, and ARCTRIG implies computing the angle, choosing either $\sin^{-1}$ or $\cos^{-1}$ so as to yield maximum accuracy.

5.6.15.3 Determination of RR Gyro Commands during RR Target Designation

When the RR is being designated towards the CSM, the commands issued to the RR CDU's for digital to analog conversion before being sent to the RR gyros are computed approximately every 0.5 seconds as follows: Let $C_S(1)$ and $C_T(1)$ be the shaft (S) and trunnion (T) commands to the RR CDU's for the antenna in Mode 1, and $C_S(2)$ and $C_T(2)$ be the corresponding commands for the antenna in Mode 2. Initially, the unit vector $\underline{u}_D$ defining the desired direction of designate in navigation base coordinates is obtained from stable member coordinates as follows:

$$\underline{u}_D = [SMNB] \text{ UNIT } (\underline{r}'_{LOS}) \tag{6.15.3}$$

where [SMNB] is defined in Section 5.6.3.2.1 and $\underline{r}'_{LOS}$ is the lag compensated range vector in Section 5.2.4.1. The commands are then computed as follows:

$$C_S(1) = K \left[\underline{u}_D \cdot \begin{pmatrix} \cos S \\ 0 \\ -\sin S \end{pmatrix} \right] \tag{6.15.4}$$

$$C_S(2) = -C_S(1) \tag{6.15.5}$$

$$C_T(1) = C_T(2) = -K \left[\underline{u}_D \cdot \begin{pmatrix} \sin T \quad \sin S \\ \cos T \\ \sin T \quad \cos S \end{pmatrix} \right] \tag{6.15.6}$$

where S and T are the present shaft and trunnion angles and K is a scale factor to establish the proper number of bits in the RR CDU's. A limit check is made to insure that no more than 384 bits are sent to an RR CDU.

5.6.16 Delta V Programs

The purpose of the Target Delta V Program (P70) is to update the estimated CSM state vector in accordance with the maneuver ΔV which is voice-linked to the LM from the CSM and then entered into the LGC as described in Sections 5.2.1 and 5.2.4.2.

The purpose of the Impulsive Delta V Program (P77) is to update the estimated LM state vector after the execution of a thrusting LM-maneuver not monitored by the GNCS.

The logic for these programs is shown in Figure 6.16-1. In the figure, $\underline{\Delta V}$ is the vehicle velocity change, expressed in that vehicle's local vertical coordinate system; and $t_{\Delta V}$ is the time of the maneuver.

START

Inhibit Rendezvous Navigation Data Processing

Inhibit Other Program Selection via V37

Extrapolate vehicle state vector to Time $t_{\Delta V}$ using Coasting Integration Routine (Section 5.2.2)

Rectify vehicle State Vector (Fig. 2.2-2) Obtaining $\underline{r}_{con}$, $\underline{v}_{con}$, $t_{\Delta V}$

$$\underline{u}_2 = -\text{UNIT}\left(\underline{r}_{con}\right)$$

$$\underline{u}_1 = \text{UNIT}\left(\underline{v}_{con} \times \underline{r}_{con}\right)$$

$$\underline{u}_0 = \underline{u}_1 \times \underline{u}_2$$

$$\underline{v}^T_{con} = \underline{v}^T_{con} + \Delta \underline{V}^T \begin{pmatrix} \underline{u}_0{}^T \\ \underline{u}_1{}^T \\ \underline{u}_2{}^T \end{pmatrix}$$

Extrapolate vehicle State Vector to original LM State Vector Time using Coasting Integration Routine

Rectify vehicle State Vector (Fig. 2.2-2) Obtaining $\underline{r}_{con}$, $\underline{v}_{con}$, $t_{\Delta V}$

Allow Other Program Selection via V37

EXIT

Figure 6.16-1. Delta V Programs: Target Delta V (P76), and Impulsive Delta V (P77)

5.6.17 ORBITAL PARAMETERS DISPLAY ROUTINE

The Orbital Parameters Display Routine R-30 may be called by the astronaut via an extended verb in order to compute and display certain orbital parameters defined below. This display will be automatically updated only when Average G is running. Also the option to select the vehicle for which these orbit parameters are to be displayed will only be available when Average G is not running.

In the normal case the apocenter altitude, pericenter altitude and the time from a reference altitude (300,000 ft for Earth orbit, 35,000 for Lunar orbit) is displayed. If the chosen orbit does not intercept this altitude, the third display is -59B59. Under certain circumstances explained in Section 4 an additional display of time from pericenter may also be requested by the astronaut. The details of each option and acceptable astronaut responses are discussed in Section 4.

The computational logic and equations used in R-30 are contained in Figs. 6.17-1 through 6.17-3. The following is a list of important parameter definitions which apply to these figures.

$\left.\begin{matrix}\underline{r}\\ \underline{v}\end{matrix}\right\}$: State vector of the selected vehicle

P_c : Primary body designator $\begin{cases}0 & \text{Earth}\\ 1 & \text{Moon}\end{cases}$

h_a : Apocenter altitude

h_p : Pericenter altitude

r_{LP} : Earth launch pad radius

r_{LS} : Lunar landing site radius

t_{ff} : Time from a reference altitude

t_{PER} : Time from pericenter

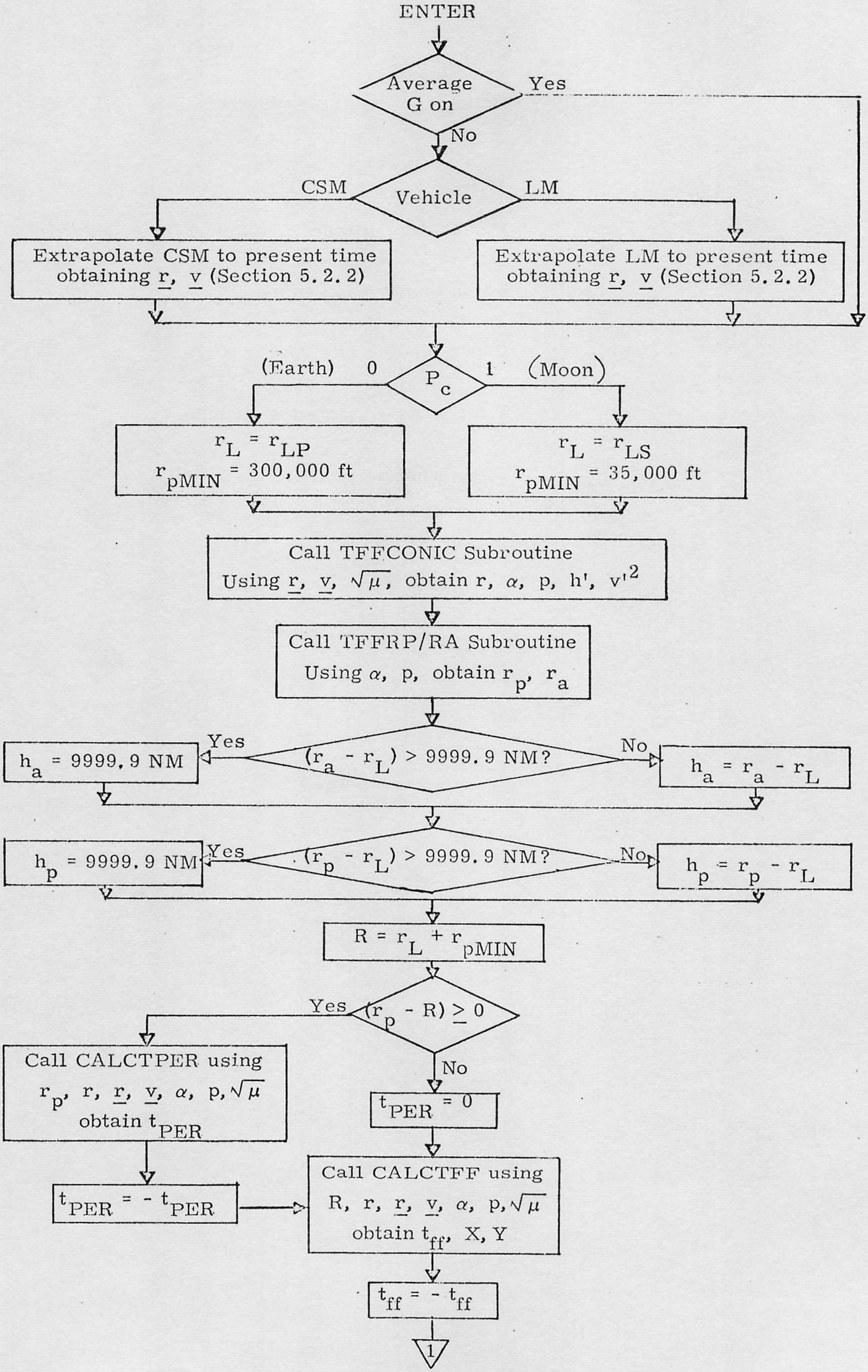


Figure 6.17-1 Orbital Parameters Display Routine
(page 1 of 2)

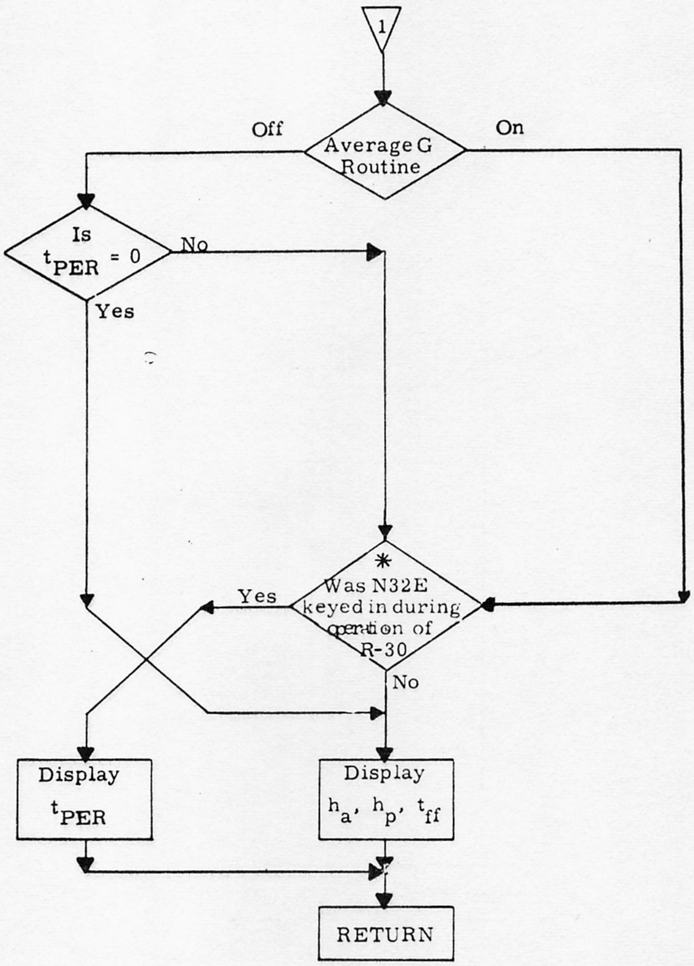


Figure 6.17-1 Orbital Parameters Display Routine
(page 2 of 2)

Note: The values of t_{ff} and t_{PER} are actually made to "count down" every second, i.e. they are automatically updated once a second, if and only if AVERAGE-G is off, although this is not shown explicitly in the above Figure.

* For full explanation of displays and DSKY options, see R30 description in Section 4 GSOP.

ENTER

$\underline{r}, \ \underline{v}, \ \sqrt{\mu}$

$$
\begin{aligned}
r &= |\underline{r}| \\
\underline{v}' &= \underline{v}/\sqrt{\mu} \\
\underline{h}' &= \underline{r} \times \underline{v}' \\
p &= \underline{h}' \cdot \underline{h}' \\
h' &= \sqrt{p} \\
(v')^2 &= \underline{v}' \cdot \underline{v}' \\
\alpha &= \frac{2}{r} - (v')^2
\end{aligned}
$$

Return $r, \ \alpha, \ p, \ h', \ (v')^2$

Figure 6.17-2a TFFCONIC Subroutine

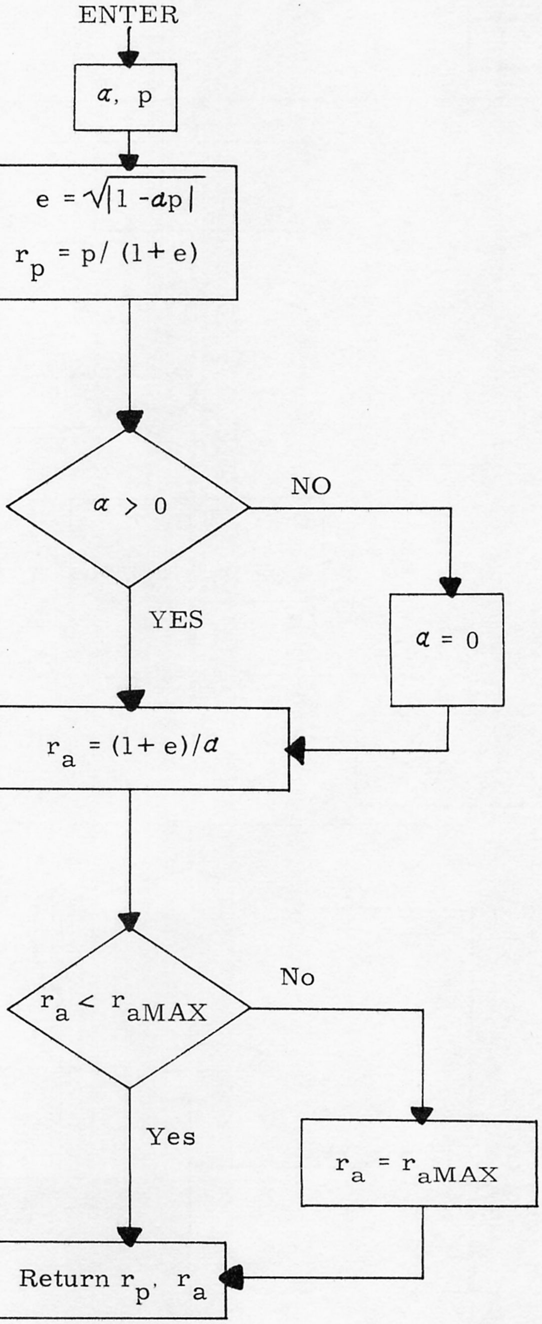
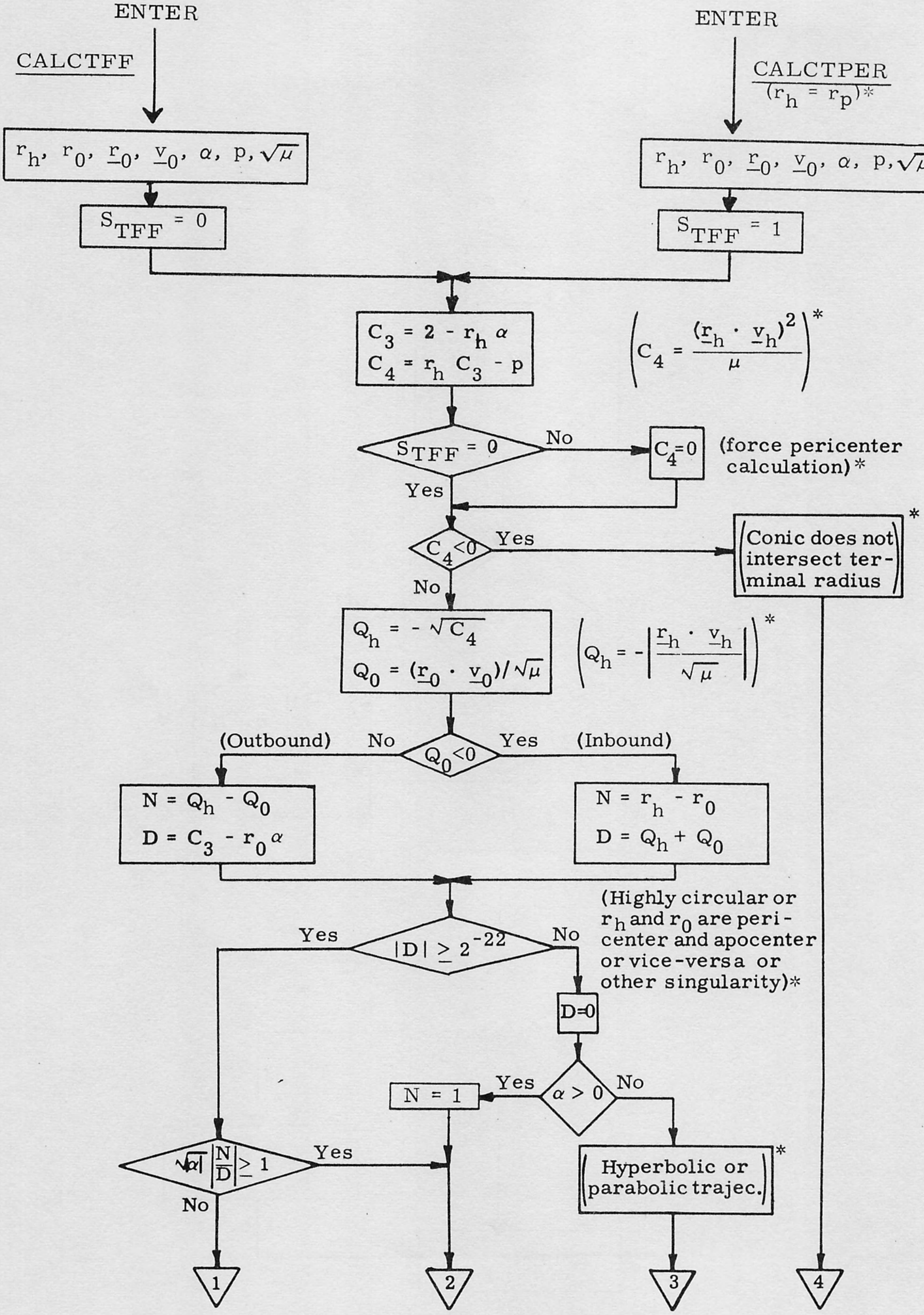


Figure 6.17-2b TFFRP/RA Subroutine



* Supplementary Information

Fig. 6.17-3 CALCTFF/CALCTPER Subroutine
(page 1 of 2)

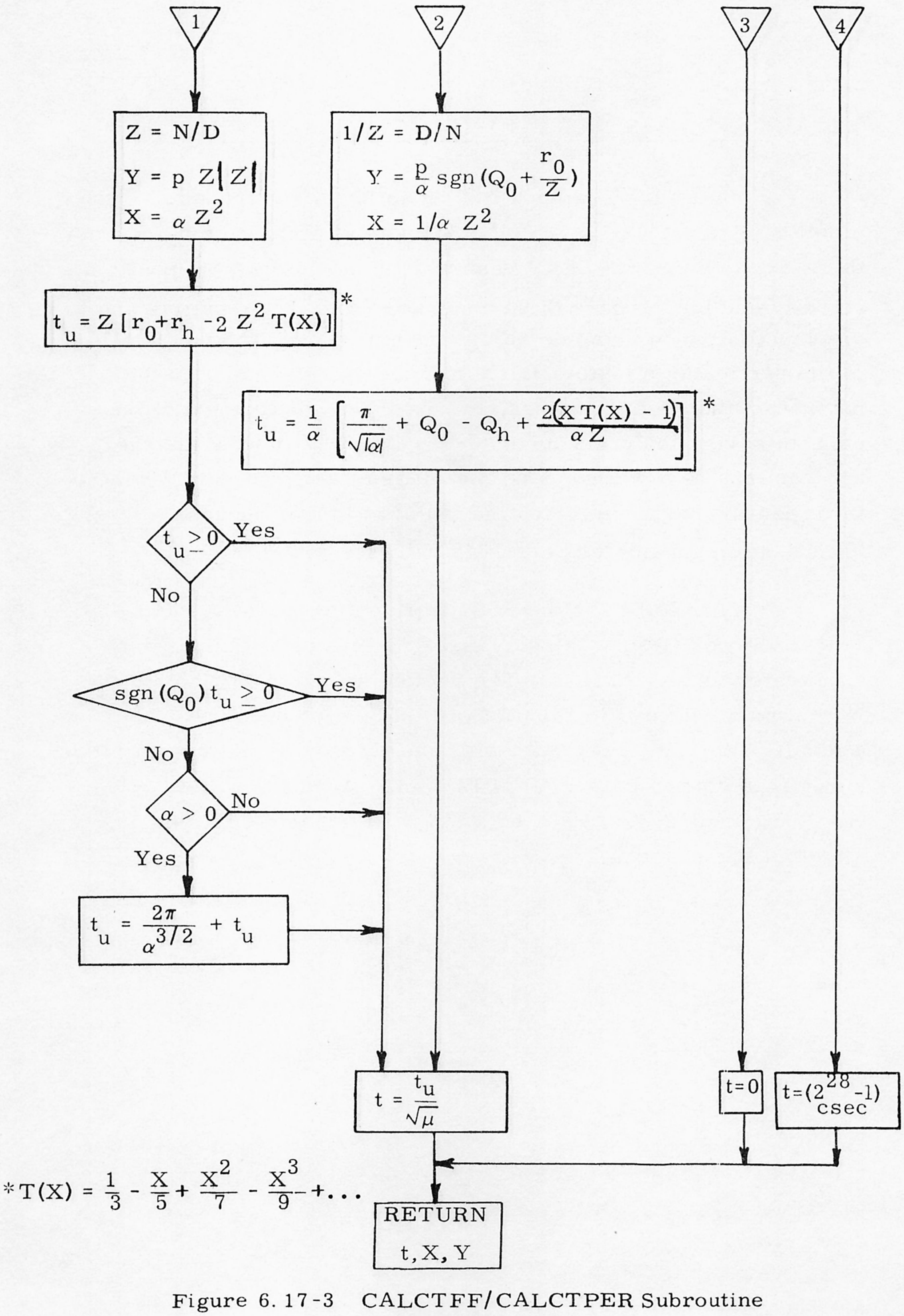


Figure 6.17-3 CALCTFF/CALCTPER Subroutine
(page 2 of 2)

5.6.19 RSS POSITION AND VELOCITY ERROR DISPLAY

In order to provide the capability for astronaut monitoring of the G&N system's estimate of state vector accuracy, there exists a special DSKY verb which causes the RSS position, velocity and bias errors to be computed from the W matrix and to be displayed. Based upon the values in this display and the details of the particular mission, the astronaut will elect to stop the navigation that is in progress, to resume or continue with the current navigation procedure, or to reinitialize the W matrix and continue navigating. The capability of selecting the W matrix initialization parameters is also included in this process. These initialization parameters are RMS values.

The logic for the RSS position error (Δr_{RSS}) and RSS velocity error (Δv_{RSS}) display and RSS Bias error Δb_{RSS} is illustrated in Fig. 6.19-1. The vectors $\underline{w}_i$ are partitions of the W matrix as defined in Eq. (2.2.19) of Section 5.2.2.4. The variables w_{rr}, w_{rv}, w_{θ}, w_{β}, $w_{\ell r}$ and $w_{\ell v}$ are W matrix initialization parameters, and RENDWFLG is the W matrix validity flag. See Sections 5.2.4.2.2 and 5.2.5.4 for further definitions and usage of these terms.

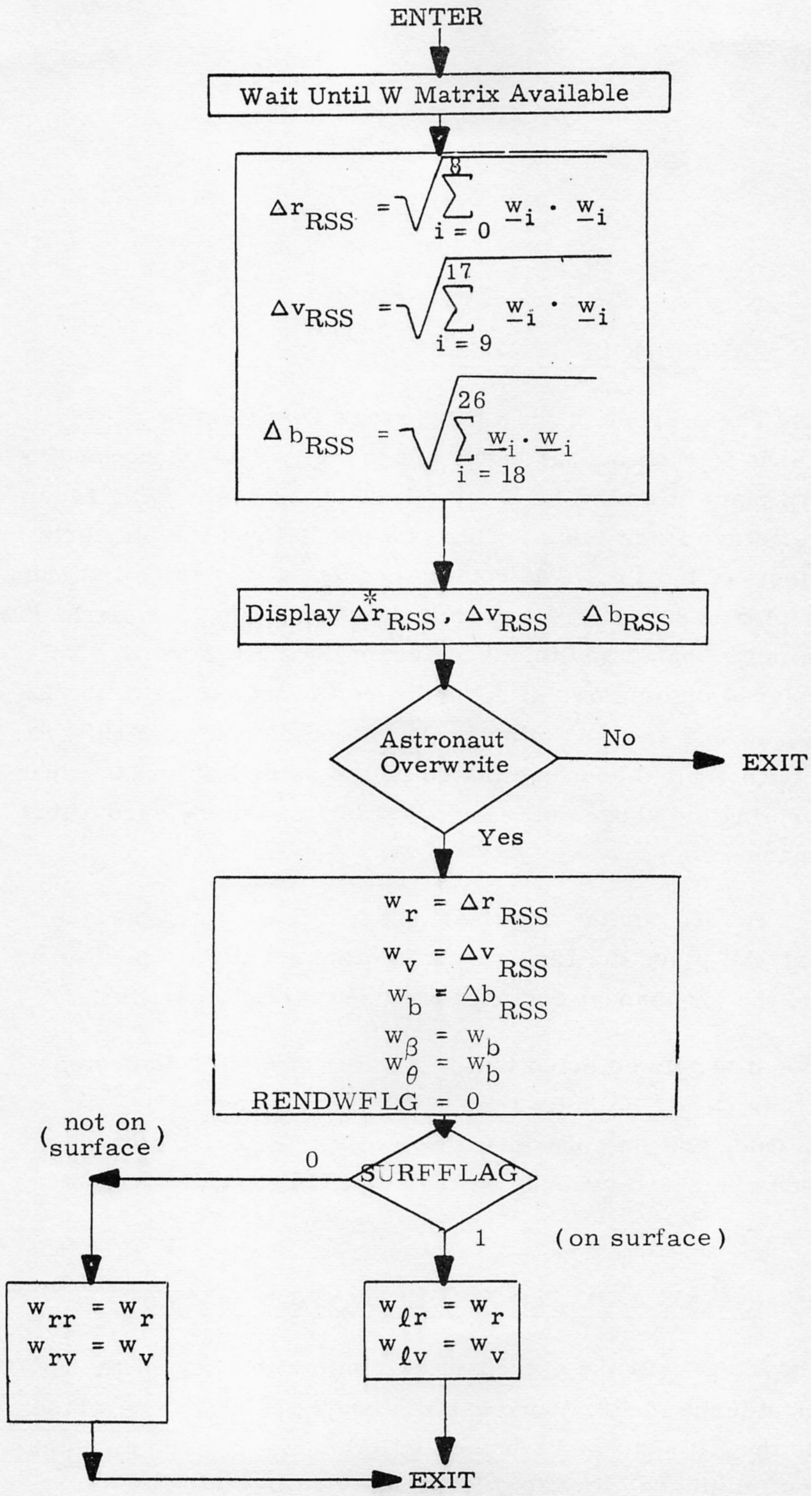


Figure 6.19-1 RSS Position and Velocity Error Display Logic Diagram

* Largest value displayed will be 99997 ft. for $\Delta r_{RSS} > 30479$ meters

5.6.20 LR SPURIOUS TEST ROUTINE

The purpose of the LR Spurious Test Routine (R-77 of Section 4) is to read out the range and velocity data of the landing radar and place it on the LGC down-link during the landing radar spurious return flight tests. This is a special routine for earth orbital tests of the LR. The routine sequentially obtains individual data samples (at 80.001 milliseconds duration each) from each of the three velocity beams and the range beam once per second. This data is placed on the down-link regardless of the status of the LR Range Data Good and LR Velocity Data Good discretes. This routine is instigated and terminated by the astronaut and is capable of performing the above data readout prior to, during, and after a DPS maneuver.

Before performing the flight test it assumed that the astronaut will place the LR antenna in either of its two positions by use of the LR manual controls provided for this purpose.

It should be noted that the status of all LR discretes (Range Data Good, Velocity Data Good, Range Low Scale, Antenna Position One, and Antenna Position Two) is already indicated at least once every two seconds on the down-link list.

5.6.21 RR LOS AZIMUTH AND ELEVATION DISPLAY

To permit the astronaut to monitor the LOS of the RR in either antenna mode, there exists a special DSKY verb which converts the RR shaft and trunnion CDU angles to more meaningful angles (azimuth and elevation) for display on the DSKY.

The azimuth and elevation angles are shown in Fig. 6.21-1 where the azimuth angle (AZ) is defined as the angle between the RR antenna LOS and the X-Z navigation base plane. The azimuth angle changes from 0 to +90 degrees as the LOS moves from the X-Z plane toward the $+Y_{NB}$ axis and changes

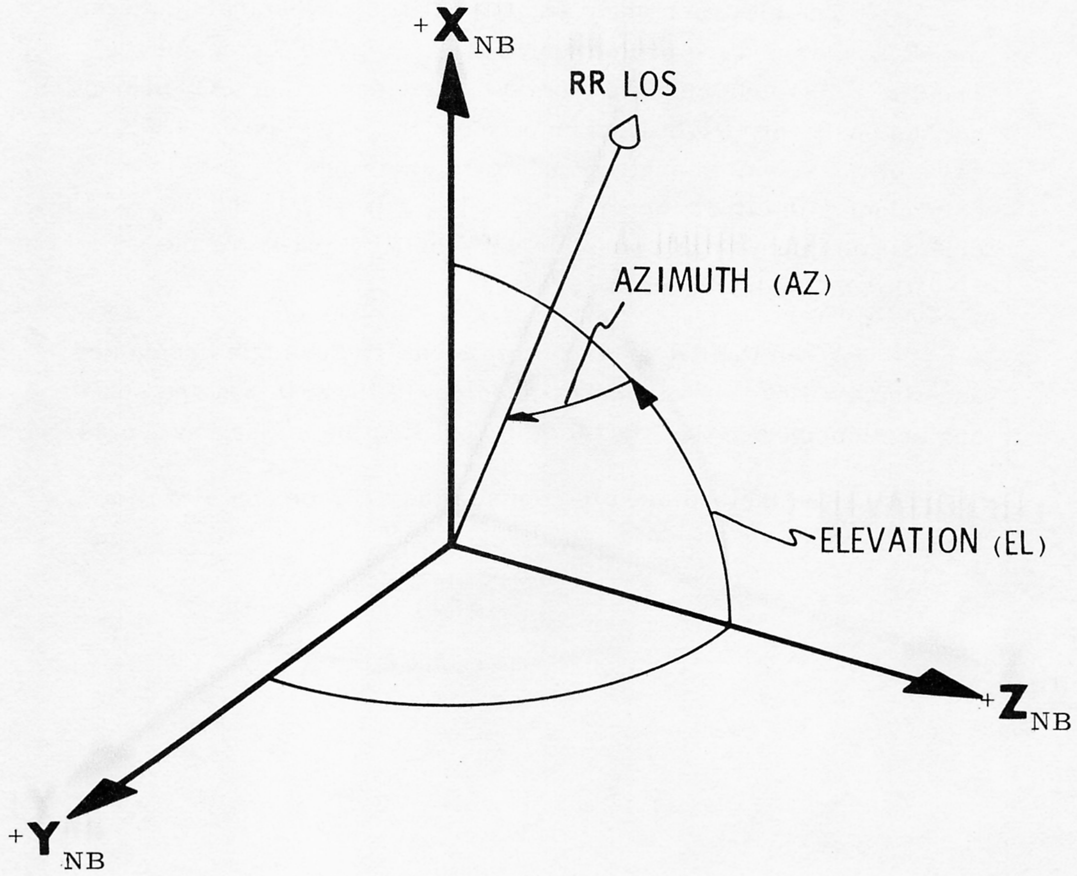


Figure 6.21-1 RR LOS Azimuth and Elevation Angles.

from 0 (360) to +270 deg as the LOS moves from the X-Z plane toward the $-Y_{NB}$ axis.

The elevation angle (EL) is defined as the angle between the $+Z_{NB}$ axis and the projection of the LOS in the X-Z plane. The elevation angle changes from 0 to +360 deg as the LOS projection rotates positively (right hand rule) about the $+Y_{NB}$ axis. The value of the elevation angle is indeterminate when the LOS is coincident with either the $+Y_{NB}$ or $-Y_{NB}$ axis. Only the $+Y_{NB}$ orientation is physically possible with the RR, and there the elevation angle displayed is +90 deg.

When the RR antenna is in Mode 1, the angles computed and displayed by this verb are equivalent to the trunnion and shaft angles displayed in the RR/LR Self Test Routine of Section 5.6.14.

The method used to compute the azimuth and elevation angles is given in Fig. 6.21-2.

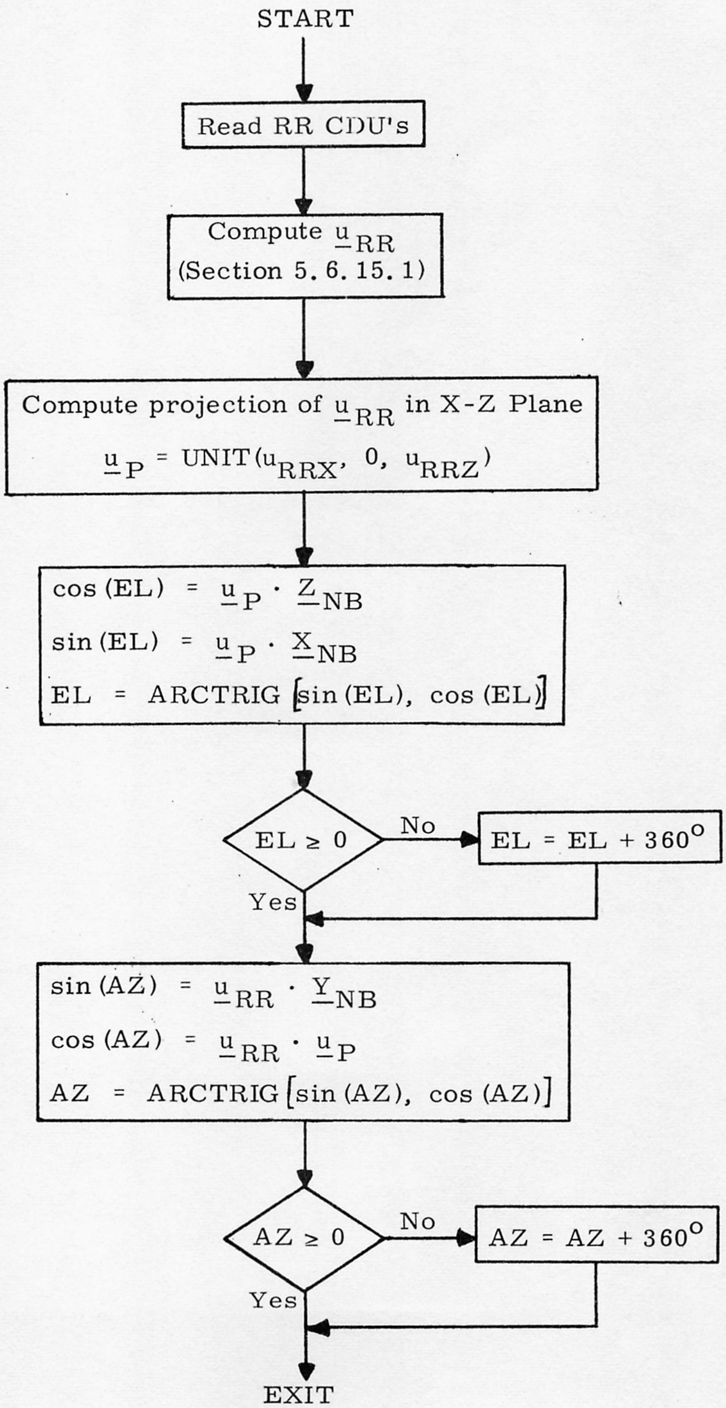


Figure 6.21-2 Computation of RR LOS Azimuth and Elevation Angles.

5.7 ERASABLE MEMORY PARAMETER LIST

This section presents a list of selected parameters required for various program operations. It should be noted that this is a very limited LGC erasable parameter list. The objective of this selected list is to identify those parameters that should be stored in erasable memory and are required primarily to initialize the operation of programs and routines. In most cases these parameters cannot be originated within the LGC and must be stored prior to the mission. Some parameters then vary continually throughout the mission (e.g. vehicle state vectors), others are constant for any one mission phase, but may vary between different mission phases, and finally some may be constant for one mission, but be required to change for subsequent missions that use the same LGC program.

Section No.	GSOP Name	LGC Name
5.2.2.3	J_{22}	
	C_{31}	
5.2.2.6	$\underline{r}_{C0}$, $\underline{r}_{L0}$	
	$\underline{v}_{C0}$, $\underline{v}_{L0}$	
	$\underline{r}_{C\ con}$, $\underline{r}_{L\ con}$	
	$\underline{v}_{C\ con}$, $\underline{v}_{L\ con}$	
	$\underline{\delta}_{C}$, $\underline{\delta}_{L}$	
	$\underline{\nu}_{C}$, $\underline{\nu}_{L}$	
	t_C, t_L	
	τ_C, τ_L	
	x_C, x_L	
	P_C, P_L	

Section No.	GSOP Name	LGC Name
5.2.4.2.2	$\delta\beta$	
	$\delta\theta$	
	0	
	var_R	
	var_{Rmin}	
	var_V	
	var_{Vmin}	
	var_β	
	var_θ	
	δr_{MAX}	
	δv_{MAX}	
	w_{rr}	
	w_{rv}	
	w_β	
	w_θ	
5.2.5.4	$w_{\ell r}$	
	$w_{\ell v}$	
5.3.2	$\Delta\underline{v}(\Delta t)$	
5.3.3.3.1	$\Delta\underline{V}_{LV}$	
	t_{IG}	
	m	
	S_E	
5.3.3.3.2	S_R	
5.3.3.3.3	$\Delta t_{Tail-off}$	
5.3.3.5	U_T	

Section No.	GSOP Name	LGC Name
5.3.4.3	k_h	LRWH
	k_{hl}	LRWH1
	LRWVX	LRWVX
	LRWVY	LRWVY
	LRWVZ	LRWVZ
	DELQFIX	DELQFIX
	q_{SW}	RPCRTQSW
	t_{SW}	RPCRTIME
	LRHMAX	LRHMAX
	LRVF	LRVF
	LRVMAX	LRVMAX
	LRWVFF	LRWVFF
	LRWVFX	LRWVFX
	LRWVFY	LRWVFY
	LRWVFZ	LRWVFZ
	$ABSC_N$ (N=0 – 4)	ABSC0 - ABSCn
	$SLOPE_N$ (N=0 – 4)	SLOPE0 - SLOPEn
	VELBIAS	VELBIAS

Section No.	GSOP Name	LGC Name
5.3.4.6 (Note 1)	m	MASS
	t_{LAND}	TLAND
	$\underline{\text{D}}\text{LAND}$	DLAND
	$\underline{r}_{LSL}$	RLS
	r_{IGZG}	RIGNZ
	r_{IGXG}	RIGNX
	v_{IGG}	VIGN
	k_X	KIGNX/B4≀
	k_Y	KIGNY/B8≀
	k_V	KIGNV/B4≀
	DELTTRIM	ZOOMTIME
	TTT^*_{CGPI} (BRAK)	TCGIBRAK≀
	TTT^*_{CGPF} (BRAK)	TCGFBRAK≀
	TTT^*_{END} (BRAK)	TENDBRAK≀
	GAIN^* (BRAK)	GAINBRAK
	$\underline{r}^*_{TG}$ (BRAK)	RBRFG
	$\underline{v}^*_{TG}$ (BRAK)	VBRFG
	$\underline{a}^*_{TG}$ (BRAK)	ABRFG
	v_{TZG} (BRAK) (Note 3)	VBRFG*
	a_{TZG} (BRAK)	ABRFG*
	j^*_{TZG} (BRAK)	JBRFG (Note 2)

(1) The landing maneuver PADLOADS share erasable locations with the Rendezvous Navigation Program P20. This places operating restrictions on P20 prior to lunar landing as described in Section 4.

(2) The LGC erasable JBRFG* is loaded with eight times the target jerk (in the units used by the descent guidance). Therefore, JBRFG* is equivalent to 8 j^*_{TZG} (BRAK), and likewise JAPFG* is equivalent to 8 j^*_{TZG} (APPR).

(3) Not referred to explicitly in GSOP.

≀ Opposite sign from GSOP notation.

Section No.	GSOP Name	LGC Name
	DELTTTAP	DELTTFAP
	TTT^*_{CGPI} (APPR)	TCGIAPPR $\wr$
	TTT^*_{CGPF} (APPR)	TCGFAPPR $\wr$
	TTT^*_{END} (APPR)	TENDAPPR $\wr$
	$GAIN^*$ (APPR)	GAINAPPR
	$\underline{r}^*_{TG}$ (APPR)	RAPFG
	$\underline{v}^*_{TG}$ (APPR)	VAPFG
	$\underline{a}^*_{TG}$ (APPR)	AAPFG
	v_{TZG} (APPR) (Note 1)	VAPFG*
	a_{TZG} (APPR)	AAPFG*
	j^*_{TZG} (APPR)	JAPFG* (Note 2)
	LEADTIME	LEADTIME $\wr$
	AZBIAS	AZBIAS
	ELBIAS	ELBIAS
	OGABIAS	OGABIAS
	AHZLIM	AHZLIM
	QHZ	QHZ
	TAUHZ	TAUHZ
	TOOFEW	TOOFEW
	2LATE466	2LATE466
	Lag/TAUROD	LAG/TAU
	MAXFORCE	MAXFORCE
	MINFORCE	MINFORCE
	RODSCALE	RODSCALE
	TAUROD	TAUROD
5.3.4.7	FLO	LOWCRIT
	FHI	HIGHCRIT

(1) Not referred to explicitly in GSOP.

(2) See Note 2, preceding page (page 5.7-4).

$\wr$ Opposite sign from GSOP notation.

Section No.	GSOP Name	LGC Name
5.3.5.3	t_{IG}	
	R_D	
	Y_D	
	$\dot{R}_D$	
	$\dot{Y}_D$	
	$\dot{Z}_D$	
5.3.5.9	R_{AMIN}	RAMIN
	$\cos(\theta_1)$	COSTHET1
	$\cos(\theta_2)$	COSTHET2

Section No.	GSOP Name	LGC Name
5.4.2.2	N	
	t_3	

Section No.	GSOP Name	LGC Name
5.4.2.4	t	
	ωt	
	$\Delta \underline{V}_{TPI}$ (LV)	
5.4.2.5	$\delta \tau_3$	
	$\delta \tau_7$	
	$\Delta \underline{V}$ (LV)	
5.4.3.1.3	J_1	J1PARM
	J_2	J2PARM
	K_1	K1PARM
	K_2	K2PARM
	θ_c	THETCRIT
	ABTRDOT	ABTRDOT
	YLIM	YLIM
5.5.2	A_X	
	A_Y	
	t_0	
	$\underline{\ell}_M$	
5.6.3	AZ_1 thru AZ_6	
	EL_1 thru EL_6	
	AZ	
	EL	

Section No.		GSOP Name	LGC Name
5.6.13		SFE_1	PIPASCFX
		SFE_2	PIPASCFY
		SFE_3	PIPASCFZ
		$BIAS_1$	PBIASX
	$\underline{B}IAS$	$BIAS_2$	PBIASY
		$BIAS_3$	PBIASZ
		ADIAX	ADIAX
		ADIAY	ADIAY
		ADIAZ	ADIAZ
		ADSRAX	ADSRAX
		ADSRAY	ADSRAY
		ADSRAZ	ADSRAZ
		NBDX	NBDX
		NBDY	NBDY
		NBDZ	NBDZ

5.8 FIXED MEMORY CONSTANTS

Section 5.8.1 contains a list and the numerical values of those fixed memory constants in Sections 5.2 through 5.6 which have not been specified previously. Those constants which are considered to be control type data are indicated by source references which are listed in Section 5.8.2. Explanatory comments are noted in Section 5.8.1 where applicable and listed in Section 5.8.3. It should be noted that only one section number is given for a constant in the list of Section 5.8.1, even though the constant may appear in other parts of Section 5. In these cases, the same value is used for the constant as reported in Section 5.8.1.

5.8.1 FIXED CONSTANTS

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.2.2.1	μ_P { μ_E		m^3/s^2	0.3986032×10^{15}	2, 3, 4	
	μ_M		m^3/s^2	0.4902778×10^{13}	2, 3, 4	
5.2.2.3	J_{2E}		–	0.10823×10^{-2}	2, 3, 4	
	J_{3E}		–	-0.23×10^{-5}	2, 3, 4	
	J_{4E}		–	-0.18×10^{-5}	2, 3, 4	
	r_E		m	6, 378, 165	2, 3, 4	
	J_{2M}		–	0.207108×10^{-3}	2, 3, 4	
	J_{3M}		–	$-2.1(10)^{-5}$	2, 3, 4	
	J_{4M}		–	0	2, 3, 4	
	r_M		m	1, 738, 090	2, 3, 4	
5.2.2.6	ϵ_t		csec	3.0	–	1
5.2.4.2.1	f_{BRR}		cps	212, 500	14	
	$f_{BRR}\tau_{RR}$		–	17, 000	5	
	k_{RR}		fps/bit	-0.6278	5, 14, 15	
	k_{R1}		ft/bit	75.04	5, 14, 15	
	k_{R2}		ft/bit	9.38	5, 14, 15	

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.2.4.2.2	var_{IMU}		$(mr)^2$	1.0	6	
5.2.5.3	ω_M		rad/sec	$2.66169948 \times 10^{-6}$	2,3,4	
5.3.2	J			1.62346×10^{-3}	2,3,4	
5.3.3.3.1	F_{DPS}		pounds	9817.5	29,30	24
	F_{APS}		pounds	3500	5,17	
	F_{RCS}		pounds	400 or 200	5,17	
5.3.3.3.3	$\Delta V_K(DPS)$ LM only		cm/sec	36	19	2
	$\Delta V_K(DPS)$ CSM docked		cm/sec	12	19	2
	$\Delta V_K(APS)$		cm/sec	308	19	2
	$\dot{m}(APS)$		lb/sec	11.32	30	
	K_1		kg-m/cs	140.12	34	
	K_2		kg-m/cs	31.138	18	
	K_3		$kg\text{-}m/cs^2$	1.5569	18	
	K_4		pounds	1050	5,17	
	$V_e(DPS)$	DPSVEX	m/sec	2955.889	29,30	24
	$V_e(APS)$	APSVEX	m/sec	3030	30	25
	F_L		pounds	400	17	

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5. 8. 2)	Comments (Sec. 5. 8. 3)
5. 3. 3. 4	$\Delta t_{Tail-off}(DPS)$		csec	-38	19, 26	
	$\Delta t_{Tail-off}(APS)$		csec	-17. 65[1)]	31	
	$\Delta t_{Tail-off}(P70)$		csec	-23. 68[2)]	32	
5. 3. 4. 2	f_B		cps	153, 600	14, 16	
	$f_B \tau_{LR}$		–	12, 288. 2	14, 16	
	k_{XA}		fps/bit	-0. 6440	14, 16	
	k_{YA}		fps/bit	1. 212	14, 16	
	k_{ZA}		fps/bit	0. 8668	14, 16	
	k_{LR2}		ft/bit	1. 079	14, 16	
5. 3. 4. 3	$\underline{u}_{RBB1}$	HBEAMNB		-0. 7147168647 -0. 0731086602 -0. 6955824372		
	$\underline{u}_{ZAB1}$	VZBEAMNB		-0. 4067366430 +0. 0954915028 +0. 9085409600		

1) Rounded off to -18 in the AGC.

2) Rounded off to -24 in the AGC.

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.3.4.3 (cont.)	$\underline{u}_{YAB1}$	VYBEAMNB		+0.0 +0.9945218954 -0.1045284632		
	$\underline{u}_{XAB1}$	VXBEAMNB		+0.9135454576 +0.0425155563 +0.4045084971		
	$\underline{u}_{RBB2}$			-0.9374036070 -0.0364014993 -0.3463371311		
	$\underline{u}_{ZAB2}$			+0.0 +0.1045284632 +0.9945218954		
	$\underline{u}_{YAB2}$			+0.0 +0.9945218954 -0.1045284632		
	$\underline{u}_{XAB2}$			+0.9999999999 +0.0 +0.0		
	Δt		sec	2		8

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.3.4.6	GUIDDURN	GUIDDURN	sec	664.40		23
	AF_{TRIM}	TRIMACCL	m/sec^2	.350133		14
	$\Delta\Delta T_{REDES}$	N/A	sec	4		Note (11a, 1) of the Guidance-and-Control Routine
	K_a	AZEACH	rad	0.01746	21, 22, 23, 35	9
	K_e	ELEACH	rad	0.01746	21, 22, 23, 35	9
	u_{XCRIT}	DEPRCRIT	rad	-.02		22
	GHZ	GHZ	m/sec^2	1.62292		
	FRAC	HIGHESTF	newtons	43,455	24	17
	qmax	PROJMAX	–	.42262	23	10
	qmin	PROJMIN	–	.25882	23	10
	RIMUZ	RIMUZ	m	1.2667		
5.3.4.7	BITPERF	SCALEFAC	bits/newton	.079796	24	17
	FMAX	FMAXPOS	newtons	43,455	24	17
	FSAT	FMAXODD	newtons	48,145	24	17
	FEXT	FEXTRA	newtons	51,331		19
	FRATE	N/A	newtons/sec	40,102	24	17
	τ_{th}	THROTLAG	sec	0.08	33	18
5.3.5.4	a_T(APS)		m/sec^2	3.2883		12
	τ (APS)		sec	919.02		12
	$1/\Delta V$		sec/m	.152		13

Section No.	GSOP Name	AGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.3.5.9	ΔV_{min}		m/sec	3.56		26
	A_T (RCS)		m/sec^2	0.785		16
5.4.3.1.2	$\dot{m}$ (DPS)		lb/sec	32.62	29, 30	24
	K (1/DV)		newtons	43670	27	27
5.5.1.2	(Kepler) ϵ_t		–	2^{-22}	12	
	(Lambert) ϵ_t		–	2^{-19}	12	
	ϵ_x (earth)		$m^{1/2}$	2^{-12}	12	
	ϵ_x (moon)		$m^{1/2}$	2^{-13}	12	
	ϵ_c		–	2^{-23}	12	
	k_1		–	2^{-17}	12	
5.5.2	A_{Z0}		rad	$4.85898502016 \times 10^{+0}$		
	ω_E		rad/U-sec	$7.29211514667 \times 10^{-5}$	8	
	B_0		rad	$4.09157363336 \times 10^{-1}$	7, 8	3
	Ω_{I0}		rad	+5.52185714700	7, 8	3
	F_0		rad	$+4.11720655556 \times 10^{+0}$	7, 8	3
	$\dot{B}$		rad/U-sec	$-7.19758599677 \times 10^{-14}$	7, 8	3
	$\dot{\Omega}_I$		rad/U-sec	$-1.07047013100 \times 10^{-8}$	7, 8	3

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.5.2	$\dot{F}$		rad/U-sec	2.67240425480	7,8	3
	C_I		–	$9.996417320 \times 10^{-1}$	7,8	3
	S_I		–	$2.676579050 \times 10^{-2}$	7,8	3
5.5.3	a		m	6,378,166	2,3,4	
	b		m	6,356,784	2,3,4	
	r_{LP}		m	6,373,338	11	
5.5.4	K_1	K1	–	0.917456380	9,10	4
	K_2	K2	–	-0.035679339	9,10	4
	K_3	K3	–	0.397836387	9,10	4
	K_4	K4	–	0.082280652	9,10	4
	LOM_O	LOMO	rev	0.534104635	9,10	4
	LOM_R	LOMR	rev/day	0.036600997	9,10	4
	A	AMOD	rev	0.017519236	9,10	4
	B	BMOD	rev	0.003473642	9,10	4
	$OMEGA_A$	1/27	rev/day	0.036291713	9,10	4
	$OMEGA_B$	1/32	rev/day	0.031250000	9,10	4
	$PHASE_A$	AARG	rev	0.024523430	9,10	4
	$PHASE_B$	BARG	rev	0.540564756	9,10	4

Section No.	GSOP Name	LGC Name	Units	Value	Reference (Sec. 5.8.2)	Comments (Sec. 5.8.3)
5.5.4	LON_O	LONO	rev	0.878830860	9, 10	4
	LON_R	LONR	rev/day	-0.00014720	9, 10	4
	LOS_O	LOSO	rev	0.273331484	9, 10	4
	LOS_R	LOSR	rev/day	0.002737803	9, 10	4
	C	CMOD	rev	0.005320572	9, 10	4
	$OMEGA_C$	1/365	rev/day	0.002737925	9, 10	4
	$PHASE_C$	CARG	rev	-0.011706923	9, 10	4
5.5.13	T_{SS}		sec	360	–	5
5.6.15.3	K		bits/rad	2200.16	–	6
5.8.1	Star Table			See Fig. 8-1	–	7

Catalogue No. (octal)		Star Name	Vis. Mag	X Coordinate	Y Coordinate	Z Coordinate
1	α	Andromedae (Alpheratz)	2.1	+.8747608555	+.0264603244	+.4838307948
2	β	Ceti (Diphda)	2.2	+.9342466124	+.1739271709	-.3113400137
3	γ	Cassiopeiae (Navi)	2.2	+.4773424940	+.1168141178	+.8709182540
4	α	Eridani (Achernar)	0.6	+.4918322686	+.2207092653	-.8422520048
5	α	Ursae Minoris (Polaris)	2.1	+.0128995818	+.0078096205	+.9998862988
6	θ	Eridani (Acamar)	3.4	+.5446995598	+.5317389073	-.6483349473
7	α	Ceti (Menkar)	2.8	+.7028937840	+.7078988678	+.0694227718
10	α	Persei (Mirfak)	1.9	+.4101921571	+.4989947555	+.7633784305
11	α	Tauri (Aldebaran)	1.1	+.3502769546	+.8927907521	+.2832502918
12	β	Orionis (Rigel)	0.3	+.2007345455	+.9691236271	-.1431958012
13	α	Aurigae (Capella)	0.2	+.1367280274	+.6814364033	+.7189922632
14	α	Carinae (Canopus)	-0.9	-.0616690080	+.6031289258	-.7952608559
15	α	Canis Majoris (Sirius)	-1.6	-.1824340636	+.9404062129	-.2869738089
16	α_2	Canis Minoris (Procyon)	0.5	-.4122762536	+.9063680102	+.0923326629
17	γ	Velorum (Regor)	1.9	-.3613649782	+.5745656444	-.7343634472
20	ι	Ursae Majoris (Dnoces)	3.1	-.4661511162	+.4772744503	+.7449243154
21	α	Hydrae (Alphard)	2.2	-.7745052221	+.6149043688	-.1484394758
22	α	Leonis (Regulus)	1.3	-.8610673209	+.4632386329	+.2096974909
23	β	Leonis (Denebola)	2.2	-.9657334280	+.0521664164	+.2542392790
24	γ	Corvi (Gienah)	2.8	-.9524366380	-.0597677121	-.2988181236
25	α	Crucis (Acrux)	1.6	-.4521486548	-.0495728431	-.8905639377
26	α	Virginis (Spica)	1.2	-.9168160791	-.3506241694	-.1910784362
27	η	Ursae Majoris (Alkaid)	1.9	-.5812217481	-.2911759648	+.7598669864
30	θ	Centauri (Menkent)	2.3	-.6895375091	-.4185354938	-.5910719617
31	α	Bootis (Arcturus)	0.2	-.7860186221	-.5221457573	+.3309660611
32	α	Coronae Borealis (Alphecca)	2.3	-.5324545035	-.7163035719	+.4510004372
33	α	Scorpii (Antares)	1.2	-.3511952476	-.8242322268	-.4441881743
34	α	Trianguli Austr. (Atria)	1.9	-.1142900725	-.3400201762	-.9334474056
35	α	Ophiuchi (Rasalhague)	2.1	-.1120382967	-.9695442116	+.2177876068
36	α	Lyrae (Vega)	0.1	+.1219537054	-.7702168243	+.6260138474
37	σ	Sagittarii (Nunki)	2.1	+.2074286490	-.8718956797	-.4435890882
40	α	Aquilae (Altair)	0.9	+.4540867734	-.8777450759	+.1528685033
41	β	Capricorni (Dabih)	3.2	+.5524232355	-.7930716636	-.2566435348
42	α	Pavonis (Peacock)	2.1	+.3205423120	-.4434583652	-.8370169081
43	α	Cygni (Deneb)	1.3	+.4542117996	-.5390337930	+.7093195408
44	ϵ	Pegasi (Enif)	2.5	+.8141988673	-.5553601830	+.1692786800
45	α	Piscis Austr. (Fomalhaut)	1.3	+.8345006310	-.2386718657	-.4965369357

Figure 8-1. Unit Vectors of the Navigational Stars

5.8.2 REFERENCES FOR FIXED CONSTANTS

1. Project Apollo Coordinate System Standards, SE 008-001-1, Office of Manned Space Flight - Apollo Program, National Aeronautics and Space Administration, Washington, D. C., June, 1965.

2. Natural Environment and Physical Standards for the Apollo Program NASA M-DE-8020-008B, April, 1965.

3. Directory of Standard Geodetic and Geophysical Constants for Gemini and Apollo, NASA General Working Paper No. 10, 020 B, April 6, 1966.

4. Apollo Missions and Navigation Systems Characteristics, Apollo Navigation Working Group Technical Report No. AN-1. 3, December 15, 1967.

5. Lunar Excursion Module Primary Guidance, Navigation and Control Subsystem, Equipment (GFE) Performance and Interface Specification, LSP-370-3, Grumman Aircraft Engineering Corporation, (C).

6. Primary Guidance, Navigation, and Control System for Lunar Excursion Module, Part I, MEI No. 6015000, M. I. T., October 15, 1965, (C).

7. Inertial Orientation of the Moon, R. C. Hutchinson, M. I. T. Instrumentation Laboratory, Report No. R-385, October, 1962.

8. Explanatory Supplement to the Astronomical Ephemeris and the American Ephemeris and Nautical Almanac, 1961.

9. JPL Ephemeris Tape E-9511.

10. Approximate Method of Sun - Moon Location, H. McOuat, M. I. T. Guidance Monitor System Note #13, December, 1965.

11. NASA letter of direction to M. I. T. 67-FS55-91-329.

12. The Fixed Memory Constants in the Conic Subroutines, W. M. Robertson, M. I. T. SGA Memo No. 8-68, March 26, 1968.

13. GAEC-MIT Interface Document, LID 280-10004, IMU-AOT-NVB-PTA Installation.

14. SCN 22-44, Specification Change Notice for LSP-370-3 (Reference 5).

15. SCN 22-50, Specification Change Notice for LSP-370-3 (Reference 5).

16. SCN 22-40, Specification Change Notice for LSP-370-3 (Reference 5).

17. Universal Mission Modular Data Book, LED-500-19, Grumman Aircraft Engineering Corporation, October 15, 1967.

18. PCR 652 Correction to Section 5 APS Minimum Impulse Burn Parameters, December 2, 1968.

19. M.I.T. Letter Number AG #521-68, 1 April 1968, also PCR 612. 2.

21. LM Landing Point Designator Procedures and Capability, J. D. Montgomery and R. H. Lefler, MSC Internal Note No. 67-EG-24, August 1, 1967.

22. Analysis of Landing Point Designator Operation, D. C. Cheatham and D. E. Steele, MSC Internal Note No. 65-EG-52, December 9, 1965.

23. A Manually Retargeted Automatic Landing System for LM, A. R. Klumpp M. I. T. Instrumentation Laboratory Report No. R-539 (Rev. 1), August, 1967.

24. PCN 765, New Propulsion System Constants.

25. PCR 253, Landing Radar Read Initiation, September 23, 1968.

26. Oral communication with Harry Byington as directed by 25th Meeting of the Apollo Software Control Board.

27. DPS Models Evaluation, R. K. M. Seto, TRW Interoffice Correspondence, 68.4711.8-187, August 22, 1968.

28. Mathematical Model for the LM Lunar Descent Hybrid Simulator, Willis M. Bolt, et. al., MSC Internal Note 68-FM-79, April 12, 1968.

29. CSM/LM Spacecraft Operational Data Book (Volume II), LED-540-54, Grumman Aircraft Engineering Corporation, June 1968.

30. Oral communication with Harry Byington of MSC on October 25, 1968.

31. GAEC TWX No. LTX-500-141, October 2, 1969.

32. PCR 846 Rev 1, More Accurate $\Delta t_{Tail-off}$ for P70, Oct. 23, 1969.

33. PCR 322 Change Fixed Constant "Throtlag" from 0.2 to 0.08.

34. PCR 1056 Rev. 1., Improvements for Impulse and Ullage Logic.

35. PCR 338, Change LPD Scaling to 1.0° in All Directions.

5.8.3 COMMENTS ON FIXED CONSTANTS

1. The quantity ϵ_t is the limit used for the minimum time step in the Coasting Integration Routine. When the time step drops below ϵ_t, the routine is terminated.

2. The ΔV_K (DPS) values and ΔV_K (APS) value are given in Reference 19. Two values are given for ΔV_K (DPS), depending on spacecraft configuration. The values were selected to be between the maximum possible ullage acceleration and minimum anticipated main engine acceleration.

3. The values given for the fixed constants used in the Planetary Inertial Orientation Subroutine (Section 5.5.2) are obtained by using Chapter 4 of Reference 7 and Sections 4C and 11D of Reference 8.

4. The Lunar and Solar Ephemerides (Section 5.5.4) are based upon the technique given in Reference 10. The fixed constants were determined from various data points on the JPL Ephemeris Tape E-9511 (Reference 9).

5. The quantity T_{SS} is a rough estimate of the time between the initiation of the Star Selection Routine in the Inflight Fine Align Routine (R-51) and the midpoint of the optical sightings on the two celestial bodies.

6. The quantity K is the scale factor used to determine the number of bits to be sent to the RR CDU's during RR target designation. This scaling is applied to the angular difference between the present and desired pointing directions of the RR. No more than 384 bits are sent to an RR CDU because of saturation conditions.

7. The direction of each of the 37 navigation stars in Fig. 8-1 is expressed as the components of a unit vector in the Basic Reference Coordinate System. These star directions are the mean places of the stars at the beginning of the Besselian year 1972. The term "mean place" is defined in Reference 8. Due to the need at M. I. T. for these star directions prior to the availability of The American Ephemeris and Nautical Almanac for 1972, the star directions were computed by M. I. T. using essentially the same technique employed by the Nautical Almanac Office.

8. The quantity Δt is the period between successive updates of the LM state vector and the guidance equations during the lunar powered descent and ascent. A value of 2 seconds was selected in order to allow the LGC sufficient time to perform all of its functions.

9. The quantities k_a and k_e are respectively the LPD azimuth and elevation increment values.

10. The quantities qmin and qmax are respectively the quantities PROJMAX and PROJMIN in Ref. 23.

12. The quantities a_t (APS) and τ (APS) are the engine performance initialization parameters estimated from LM mass and off loaded by descent RCS fuel consumption. These will be replaced by the output of the Thrust Magnitude Filter after its first pass.

13. The quantity $1/\Delta V$ is the initial value of $1/\Delta V_1$, $1/\Delta V_2$, and $1/\Delta V_3$ used in the thrust filter computations and is equal to $1/2\ a_T$ (APS) where a_t (APS) is given in Section 5.8.1.

14. AF_{TRIM} is the expected acceleration during the trim phase. It is computed as the thrust corresponding to the throttle setting of the trim phase divided by the mass expected at that time.

16. A_T(RCS) is the maximum possible acceleration available with 4 RCS jets on and the LM dry.

17. FRAC, BITPERF, FMAX, FSAT, and FRATE are non-critical constants derived from Data Exchange Index No. 2.6.3.3.1 "Engine Tag Values and Class Influence Coefficients for LM-5" dated 3/12/69, as authorized by PCN 765.

18. τth is the empirically determined time interval used for providing minimum overshoot in throttle response at throttle recovery time.

19. FEXT is the non-critical bias added to throttle commands to assure setting throttle against mechanical stop.

22. u_{XCRIT} is the arbitrary limit on the pointing of the unit line-of-sight to the landing site to assure landing site redesignations do not produce a site behind the current LM position. This constant allows a maximum forward landing site range of 50 times the current altitude.

23. GUIDDURN is the non-critical estimate of the duration from start of FTP thrust in the braking phase to touchdown.

24. The quantities F_{DPS}, V_e(DPS), and $\dot{m}$ (DPS) are the average values determined from data in References 29 and 30.

25. The quantity V_e (APS) was determined from data in Reference 30. It has been rounded in the coding to 3030 m/sec. This will not affect its use in the program.

26. The quantity ΔV_{min} is equivalent to 2 seconds of an acceleration 10 percent larger than lunar gravity.

27. The value used is based on early engine data, and is accurate enough for our purposes here. In the LGC number loaded appears as 436.70; the factor of 100 is maintained by using $\Delta t = 2$ instead of 200.

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